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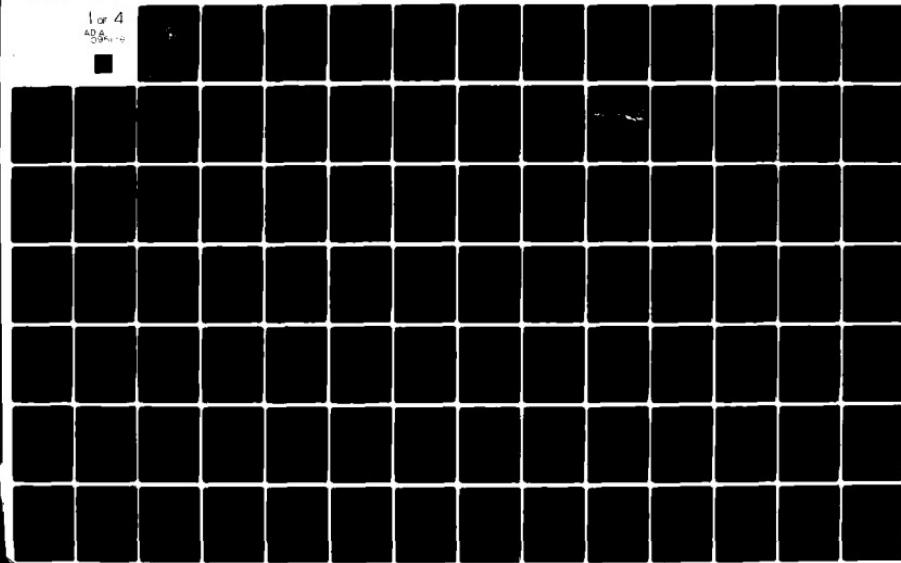
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THESIS

TACTICAL MISSILE CONCEPTUAL DESIGN

by

Danny Ray Redmon

September 1980

Thesis Advisor:

G.H. Lindsey

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Tactical Missile Conceptual Design

by

Danny Ray Redmon
Lieutenant, United States Navy
B.S., Purdue University, 1974

Submitted in partial fulfillment of the
requirements for the degree of

AERONAUTICAL ENGINEER

from the

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ABSTRACT

This thesis presents the theory necessary for the conceptual design of a tactical missile. The design process begins with the well known linear aerodynamic theory for initial sizing and later includes nonlinear effects to determine the final design of the missile. Where theory does not apply, empirical methods are presented which are known to give accurate results. An air-to-air missile is designed for a specific threat as an example which immediately follows the development of the theory for each section. Several small digital computer programs are presented and used for analysis of specific areas of the design. One large program (AERO1) is used for determining the aerodynamic coefficients of the final design.

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I. INTRODUCTION

This study was made to present a method for the conceptual design of tactical missiles. The starting point for the design was a recently completed report by General Dynamics, Convair Division entitled Rapid Approach to Missile Synthesis (RAMS). A procedure was then developed more akin to aircraft design and which bears little resemblance to RAMS and which uses basic equations to size components rather than nomograms and table look-up. The procedure starts with a threat description and proceeds with the formulation of performance objectives and a conceptual design of a tactical missile to counter the threat. The design is not unique, but as will be shown, is a compromise of parameters to give one possible solution to the design problem; therefore, the point design arrived at is not necessarily the optimum design for the presented threat. An attempt is made to find the optimum performance within specific areas of the design process.

Throughout this study the theory involved is explained and specific examples are worked. A complete design example is worked out in detail. It is an air-to-air missile designed to counter the new Soviet RAM-K fighter aircraft. This example is worked in each section immediately following the development of the theory for that specific area.

II. PROBLEM DEFINITION

A. THREAT ANALYSIS

1. Operational Requirements

The design of a new missile is usually in response to an operational requirement which arises as the result of one of the following: (1) A new technology provides the means to design a more effective missile to meet a current threat. An example of this might be an advance in material science, which allows higher inlet turbine temperatures for a turbojet engine; therefore, allowing higher missile flight speeds. (2) Intelligence indicates a new threat for which existing missiles are not effective. (3) Operational reports indicate a current missile is inadequate against a current threat.

Regardless of how the operational requirement is derived, a statement of the threat is required before the design process can proceed. Experience has proven that one missile cannot be designed to meet all types of threats without seriously compromising performance or effectiveness. This can be illustrated with the design of the warhead. A contact fuze, shaped charge warhead designed to penetrate and kill hard targets such as tanks, would not be effective against a highly maneuverable aircraft for which the expected miss distance is several feet. For this reason the design of a missile must start with a detailed analysis of the threat. The more detailed this analysis is, the more effective the final design can be.

2. Design Example (Operational Requirement)

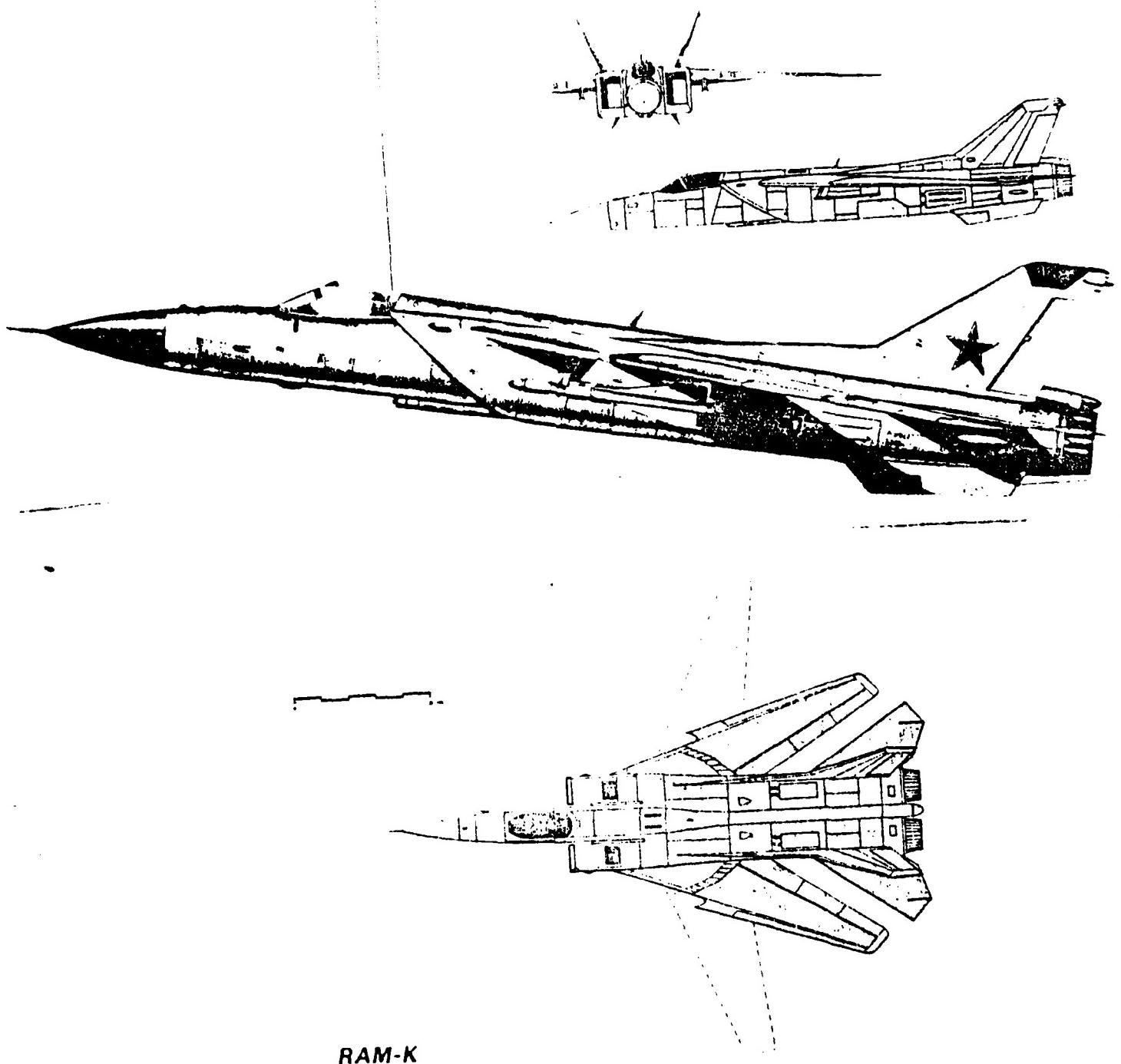
A design example will be used as a continuous thread throughout this thesis to demonstrate applications of the theory. An air-to-air missile will be designed to counter the new class of Soviet fighter, which is in the advanced development stage at the Ramenskoye Experimentation Center. The fighter, as described in Aviation Week [1,2] has been designated the RAM-K. The RAM-K is a twin engine fighter with variable geometry inlets and swing wings. The aircraft bears a resemblance to both the F-14 and F-15. It is expected to be the recipient of a new look down, shoot down radar and the 40 km range AA-X-9 missile. The following unclassified dimensions and performance data are available on the RAM-K:

Wing span	40 ft
Overall length	64 ft
Gross weight	60000 lbs
Maximum speed	M=2.5
Service ceiling	60,000 ft

Figure (2-1) is a drawing of the RAM-K.

3. Scenario

The scenario within which the missile is expected to operate should also be described. If the normal mode of operation of the threat is not known, an attempt should be made to define the most demanding scenario that can be expected. For a defensive weapon the most challenging incoming threat will normally be a head-on encounter. The threat profile may vary from a high level attack with a terminal dive to a low level attack with a



RAM-K

Figure (2-1). RAM-K Fighter.

terminal pop-up maneuver. In the case of the AS-6 (Kingfish) two modes of attack can be expected. In such a case both profiles must be evaluated to determine the most demanding in terms of missile performance objectives. For an offensive system, such as an air-to-air missile designed to intercept and destroy an enemy fighter before it launches its weapons, the scenarios analyzed should include all possible encounter geometries.

4. Design Example (Scenario)

The scenario for the above threat would likely be an intercept situation in defense of the fleet high value unit. The scenario is taken to be a head-on encounter with the missile and the target at the same altitude. Since the combat specifications of an aircraft are normally given at 10,000 ft., this is taken as the scenario altitude.

B. HISTORICAL SURVEY

Missile design is an iterative process, and the first time through the design loop many assumptions have to be made concerning component sizes and weights. One method of approach at the early stage is to employ historical data of existing missile sizes and weights; since justifications for these parameters were made during their design processes. An example of the use of historical data in determining the initial missile length can be made with the length to diameter ratio. The length to diameter ratios of existing missiles of the same class as that

being designed are collected, and an average is computed. The diameter of the design is fixed by one of three driving factors (propulsion, warhead, or guidance). From the average length to diameter ratio the initial missile length is then estimated. From this historical data, initial choices based on the experience of others can be made for many of the missile parameters. These parameters define a baseline missile, which is the initial configuration from which design iterations and refinements can be made.

Since missiles are designed for specific missions, specific parameters such as length and diameter are of little value in comparing missiles. Dimensionless ratios such as length to diameter, L/D, ratio and aspect ratio, AR, are more meaningful when relating missiles. Some parameters which are useful in defining the baseline missile are listed below:

L/D = Length to diameter ratio

L_n/D = Nose length to diameter ratio

AR_w = Aspect ratio of the wing

AR_t = Aspect ratio of the tail

W/S = Weight to lifting surface area ratio

$V_t = S_t l_t / (S_{ref} d_{ref})$ Tail volume coefficient

W_G/W_{wh} = Gross weight to warhead weight ratio

The tail volume coefficient, V_t , is a dimensionless parameter used to initially size the tail. For a tail control missile, it is a measure of the relative control effectiveness when comparing missiles. For a wing control missile it is a relative measure of stability.

TABLE 2-I
HISTORICAL DATA

Missile	L/D	L_n/D	A_{r_w}	$AR_{t/c}$	W/S	$S_{t/c}/S_w$	$V_{t/c}$	W_g/W_{wh}
ASAIM	8.00							
Phoenix	10.00	2.32	.60	1.53	134.20	.40	5.76	
Sidewinder ^C	23.75	2.00	.97	4.00	50.60	.20c	38.88	8.22
Sparrow	17.14	2.40	2.82	2.59	121.95	.60	28.80	5.72
Magic	18.82							7.17
R530	12.58		1.48	2.15	60.31	.32		7.23
Sky Flash	17.29	2.00	.97	4.00	50.60	.20	38.88	5.68
Shafrir ^C	17.00		1.36	3.48c	70.45	.21c		8.44
Aspide	18.43	2.42	2.84	2.77	110.90	.58	30.43	6.30
<u>Average</u>	15.89	2.23	1.61	2.61	88.09	.42	28.55	6.97
				3.74c		.20c		

A complete historical survey should not be limited to the parameters listed here. Any dimensionless parameter which will add information about the proposed design should be included for completeness.

Table 2-I is an example of a collection of such parameters for existing air-to-air missiles [1,3]. In this table the subscript, c , is used to indicate a canard control surface.

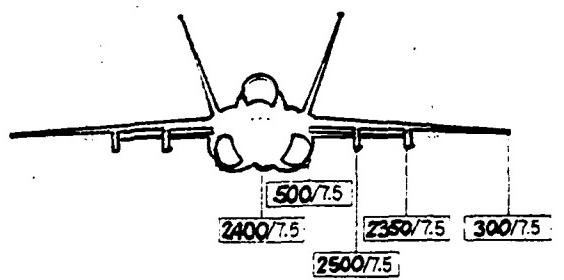
C. LAUNCH PLATFORMS AND PHYSICAL CONSTRAINTS

The problem definition must also include a description of the intended launch platform for the missile. The aircraft or shipboard launcher from which the missile will be launched will fix many design features of the missile. For instance the most important consideration in the problem definition phase is any physical constraints imposed by the launcher. Since it is not normally feasible, economically to design a launcher to fit the missile, most new missiles must fit existing launchers. In the case of shipboard launchers there will be a maximum length and diameter and a maximum launch weight which can be accommodated.

For the case of an air launched missile, there will be a maximum weight, and the dimensions may also be limited due to the performance requirements of the aircraft.

For the air-to-air missile design example of this study, the launch platforms will be the F-16 and F-18A. Figure (2-2), which is from Interavia [4,5] shows the pylon weight limitations for these aircraft. From these figures it can be seen that the maximum launch weight for this design is limited to 2500 pounds by the pylon limitations of the F-18A.

Load Carrying Capability of F-18A



Load Carrying Capability of F-16

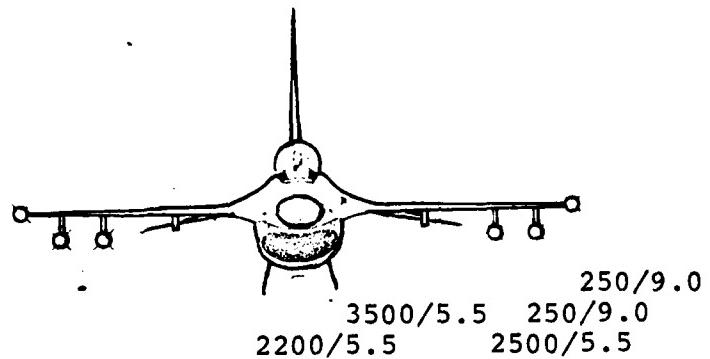


Figure (2-2). Weight Limitations.

D. MISSION PROFILES AND PERFORMANCE OBJECTIVES

The mission profile of a missile consists of dividing the flight into fundamental segments which consist of a single function, such as boost to cruise speed and altitude, cruise to the target, and terminal homing phases. The mission profile will vary from missile to missile. For a cruise missile it may consist of a series of pop-up maneuvers and low level cruises. For a short range missile, the profile may be entirely terminal homing. The Mach number and altitude are specified at the beginning and end of each mission segment, as well as the range covered by each segment. The range covered by a segment can be considered in one of two ways. If the segment is short, such as the terminal phase, the distance along the intended flight path is considered. For longer range missiles, the distance over the ground is of importance. The mission profile must be defined during the problem definition phase in order to specify missile performance objectives.

The mission profile of a missile normally consists of a boost, cruise (mid-course) and terminal phase. The boost phase accelerates the missile to its flight speed. This acceleration may be large for a surface-to-air missile which must be accelerated from rest to a high supersonic speed or it may be small for an air launched missile which has the speed advantage of the aircraft from which it is launched.

The cruise segment, or mid-course phase, primarily is used to deliver the missile to a point in space where the seeker can

acquire the target. The range and speed of the cruise segment is then a function of the required stand-off distance for the target.

The terminal phase is somewhat more difficult to analyze. If the target maintains constant heading and velocity the flight path of the missile may be modeled by circular arc segments. This method will give approximate values of range. If the target maneuvers, the missile must follow and the range and speed requirements become more complicated.

In determining the range and speed requirements, all expected encounter geometries should be analyzed. The most demanding encounter will then fix the performance objectives. The most demanding speed requirement, in terms of maintaining a minimum stand-off distance, will normally be a head-on encounter. Although the required missile speed and range are determined in this section, the missile velocity may be varied later in the design process due to guidance considerations.

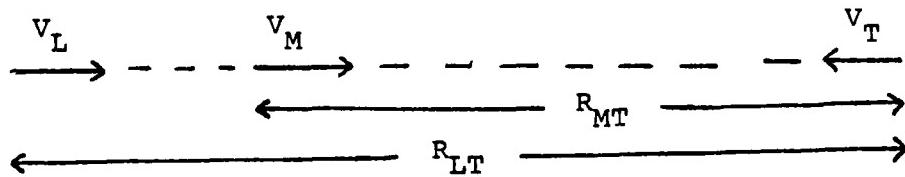
1. Design Example (Mission Profile)

From the threat defined above, the ideal situation would be to obtain a fire control solution and launch such that the minimum separation distance between the launch aircraft and the target is 40 km. This can be accomplished in one of two ways. The launch aircraft can fire a semi-active homing missile at such a range and speed that intercept occurs before the minimum range is reached, or an active homing missile can be fired, and once missile lock-on is achieved, the launch aircraft can maneuver to maintain the minimum separation distance.

The active radar homing missile would decrease the launch range and the range required of the missile, which will lessen the constraints on the launch aircraft. The terminal portion of the engagement is a function of the guidance law and will be determined in Chapter 3.

Two cases are investigated to determine the effect on the range requirement of the missile when a minimum separation distance from the target to the missile of 40 km is maintained. The first case is a semi-active homing missile, for which the launch aircraft must maintain a closing course until intercept. The second case is an active homing missile which has a lock-on range of 10 km. The launch aircraft may then maneuver to maintain a separation distance.

a. Case 1: Semi-active homing missile



R₀ = Range at which the missile is launched

M_L = 1.5 = Launch Mach number

M_M = 2.5 = Missile Cruise Mach number

a = Speed of sound

V_L = M_La = 1.5a = Launch speed

V_T = M_Ta = 2.5a = Target speed

V_M = M_Ma = 2.5a = Missile speed

The instantaneous range from the missile to the target is given by, R_{MT}

$$R_{MT} = R_0 = (V_M + V_T)t \quad (1)$$

The instantaneous range from launch aircraft to the target is given by, R_{LT}

$$R_{LT} = R_0 - (V_L + V_T)t \quad (2)$$

If the target does not maneuver, intercept will occur at t_f , when $R_{MT} = 0$

$$0 = R_0 - (V_M + V_T)t_f$$

$$t_f = \frac{R_0}{(V_M + V_T)} = \frac{R_0}{5a}$$

If the launch aircraft is at the minimum separation distance, $R_{LT} = 40$ km, when intercept occurs.

$$40 \text{ km} = R_0 - (V_L + V_T)t_f$$

substituting for V_L, V_T and t_f

$$40 \text{ km} = R_0 - (1.5 + 2.5)a\left(\frac{R_0}{5a}\right)$$

Solving for R_0

$$R_0 = 200 \text{ km} = \text{Launch range}$$

The range required of the missile is then, R_M .

$$R_M = V_M t_f = 100 \text{ km} = 53.96 \text{ nmiles}$$

If the missile speed is increased to $M_M = 3.0$, then

$$R_0 = 148.15 \text{ k and,}$$

$$R_M = 80.81 \text{ km} = 43.61 \text{ nmiles.}$$

b. Case 2: Active Homing Missile

The lock-on range is a function of the seeker in the missile and will be covered later in this thesis. If it is assumed that the launch aircraft must maintain its course until lock-on occurs at a range of R_{LO} , the problem can still be solved. The geometry is the same as in Case 1. Instead of following a constant course until intercept, the launch aircraft must now only maintain a closing course until $R_{MT} = R_{LO}$. Then from equation (1)

$$R_{MT} = R_{LO} = R_0 - (V_M + V_T) t_{f1}$$

Solving for t_{f1}

$$t_{f1} = \frac{R_0 - R_{LO}}{(V_M + V_T)}$$

If at the time of target lock-on, t_{f1} , the target and launch aircraft are at the minimum separation distance, $R_{LT} = R_{min}$, from equation (2),

$$R_{LT} = R_{min} = R_0 - (V_L + V_T) t_{fl}$$

Inserting for t_{fl} ,

$$R_{min} = R_0 - (V_L + V_T) (R_0 - R_{LO}) / (V_M + V_T)$$

Solving for R_0 ,

$$R_0 = \frac{R_{min}}{\left[1 - \left(\frac{V_L + V_T}{V_M + V_T} \right) \right]} - \frac{\left(\frac{V_L + V_T}{V_M + V_T} \right) R_{LO}}{\left[1 - \left(\frac{V_L + V_T}{V_M + V_T} \right) \right]} \quad (3)$$

For the same geometry and relative speeds of the first case, with $M_M = 3.0$,

$$t_{fl} = \frac{R_0 - R_{LO}}{5.5a}$$

A reasonable value of lock-on range is 10 km. This will be shown later in the guidance section of the study. The time to lock-on then becomes,

$$t_{fl} = \frac{R_0 - 10}{5.5a}$$

From equation (3), R_0 then becomes, $R_0 = 118.52$ km. The missile range to lock-on is then, R_{M1} ,

$$R_{M1} = V_M t_{fl} = 59.19 \text{ km} .$$

If the target does not maneuver the time from lock-on to intercept becomes, t_{f2} .

$$t_{f2} = \frac{R_{LO}}{(V_M + V_T)}$$

and the missile range from lock-on to intercept becomes,

$$R_{M2} = V_M t_{f2} = 5.45 \text{ km}$$

The total missile range is then the sum of the two,

$$R_M = R_{M1} + R_{M2} = 64.64 \text{ km} = 34.88 \text{ nmiles} .$$

As can be seen from the above analysis, both the detection range of the target and the required missile range are decreased significantly when an active homing missile is used. On the other hand, it must also be remembered that the complexity and cost of the missile will be increased as a result of choosing an active radar seeker. For the design example in this study, an active radar seeker is chosen; therefore, the maximum range requirement will be 35 nmiles at a speed of $M_M = 3.0$, however, this missile velocity is tentative until a guidance analysis is complete.

From the preceding analysis the mission profile is determined. It must be kept in mind that the mission profile may be changed during the design process to meet other design objectives. The following profile assumes both the target and launch aircraft at the same altitude.



BOOST

CRUISE

TERMINAL
HOMING

SEGMENT	M_{begin}	h_{begin}	M_{end}	h_{end}	RANGE
(1) Boost	1.5	10,000 ft	3.0	10,000 ft	
(2) Cruise	3.0	10,000 ft	3.0	10,000 ft	29.6 nmiles
(3) Terminal	3.0	10,000 ft	3.0	10,000 ft	5.4 nmiles

III. GUIDANCE LAW SELECTION

Although the specifics of the guidance system is beyond the scope of conceptual design, the selection of a guidance law is necessary for initial calculations. The warhead design depends on the expected miss distance between the missile and the target and the lifting surface area depends on the maneuvering requirements of the missile. Both the miss distance and maximum acceleration required are functions of the missile guidance law.

The guidance law for a missile is the analytical formulation used by the guidance system to convert sensed target information into missile steering commands. Three general guidance laws are used. Most others can be forced to fit into one of these categories. These are:

- 1) Pursuit Guidance
- 2) Line-of-Sight Guidance
- 3) Proportional Guidance

A. PURSUIT GUIDANCE

A pursuit guidance law is illustrated in Figure (3-1a) and is one in which the missile velocity vector is always directed toward the target. The target and the missile velocity vectors must therefore be sensed; so this type of guidance normally assumes an on-board tracker. The missile may have a separate mid-course guidance package to increase range, but target lock-on initiates the pursuit guidance for the terminal homing phase.

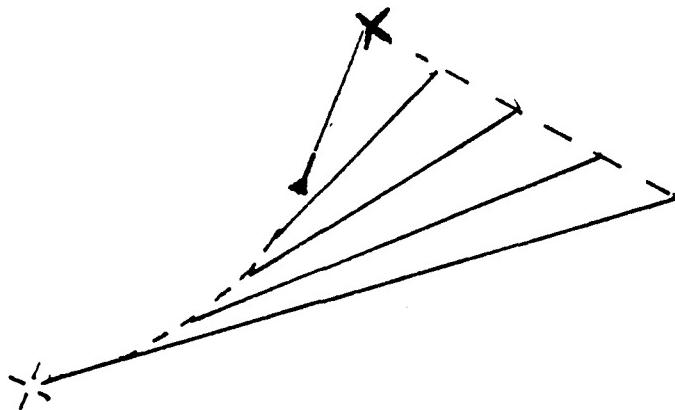


Figure (3-1a). Pursuit guidance.

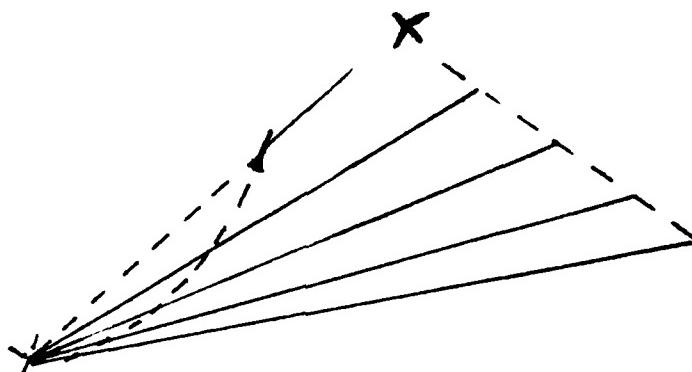


Figure (3-1b). Beam rider guidance.

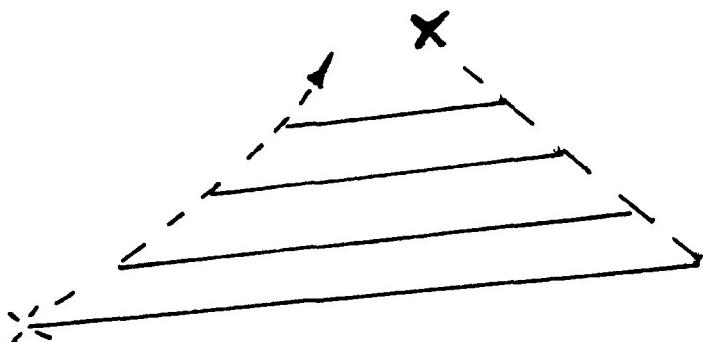


Figure (3-1c). Proportional guidance.

For this reason it has the advantage of launch-and-forget at lock-on. Since the signal processing is limited to looking and pointing, the avionics are relatively simple and usually on-board the missile. An option for this type of guidance would be to include a lead angle to accomodate faster moving targets.

B. LINE-OF-SIGHT GUIDANCE

Line-of-sight guidance is used in a beam rider missile. This guidance scheme is illustrated in Figure (3-1b) and requires that the missile remain on a line (beam) joining the target and a control point. The target tracker is located at the control point; therefore, avoiding the necessity of an on-board tracker. Because of this, a dedicated fire control system is needed from launch to intercept. The range of this type of guidance is normally less than with the other types. A speed advantage is required for line-of-sight guidance since no lead angle is incorporated. The main advantage of this type of guidance is the simple avionics required to maintain the missile in the beam.

C. PROPORTIONAL GUIDANCE

A proportional guidance law is one in which the rate of change of the missile heading is made proportional to the rate of change of the line-of-sight between the missile and the target. This is illustrated in Figure (3-1c). Since the guidance law anticipates the target's future position, it can attain a higher degree of responsiveness than other guidance laws. In proportional guidance the rate of change of the line-of-sight must be sensed

on-board the missile. Because of this requirement, and the need to provide anticipated steering commands, the avionics required are the most complex of the three guidance systems.

D. COMPARISON OF GUIDANCE LAWS

In early design considerations two parameters of interest are acceleration required of the missile and the miss distance attainable. Of the three guidance laws only the proportional law can respond to fast maneuvering targets. Since the missile must stay in the line-of-sight for a beam rider system, any target maneuver will cause large excursions in the missile flight path, resulting in large normal accelerations. The pursuit guidance law causes similar large excursions near intercept due to the velocity vector always pointing at the target.

Several system parameters affect the miss distance attainable with a particular guidance law. An excellent source on the effects of these parameters is an article written by Dr. Robert Goodstein [6]. The parameters studied for their effect on miss distance were:

- 1) Sensor Bias Angle
- 2) Noise
- 3) Target Heading
- 4) Target Acceleration
- 5) Target Speed
- 6) Wind Gusts

The results have been reproduced and are included in Figures (3-2) through (3-7). Table 3-I provides overall guidance in the selection

of a guidance law and is also reproduced from the above reference. A first glance would indicate that proportional guidance is a proper choice for all cases. It must be kept in mind, though, that cost and simplicity are also driving factors in the design process. Furthermore, it can be seen that, while proportional guidance with a high gain has good performance against maneuvering targets, any noise in the system will highly degrade this performance. For this reason another guidance law may be desired, or a compromise in the gain selection may have to be made in which some performance is given up in order to deal with a noisy system. A reasonable range of proportionality constants that gives good performance against both maneuvering targets and noisy systems is $k = 2$ to $k = 6$.

Once a guidance law is selected, a more detailed analysis has to be performed to determine if the maximum acceleration required of that particular guidance system is within the attainable maneuverability limits for the missile. The maximum acceleration required, in turn, determines the lifting surface area needed. A good estimate of maximum acceleration, which keeps the miss distance less than 50 feet, is to set it equal to three times the target acceleration plus ten.

$$a_m = 3a_t + 10$$

Figure (3-8) shows the miss distance sensitivity to target acceleration.

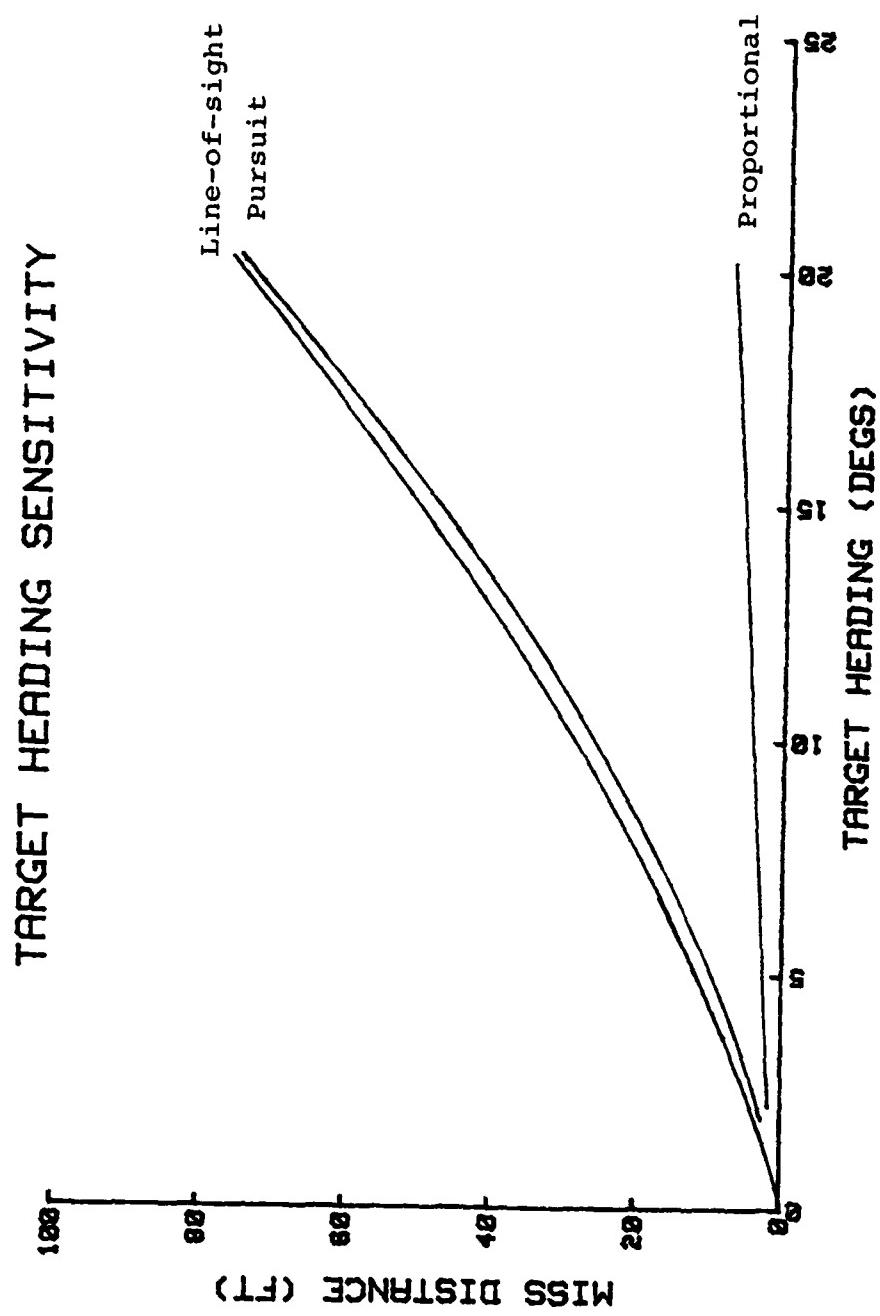


Figure (3-2)

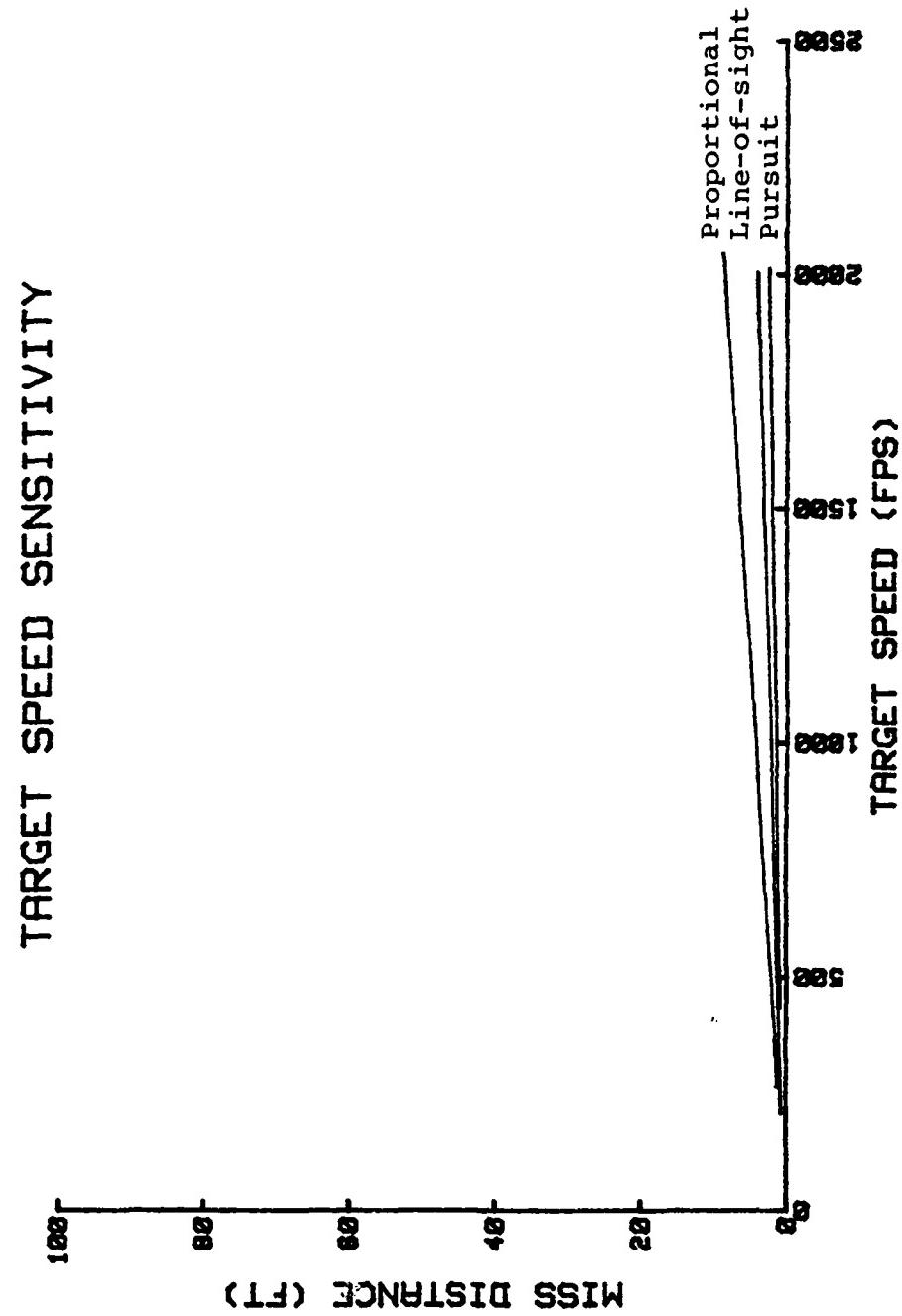


Figure (3-3)

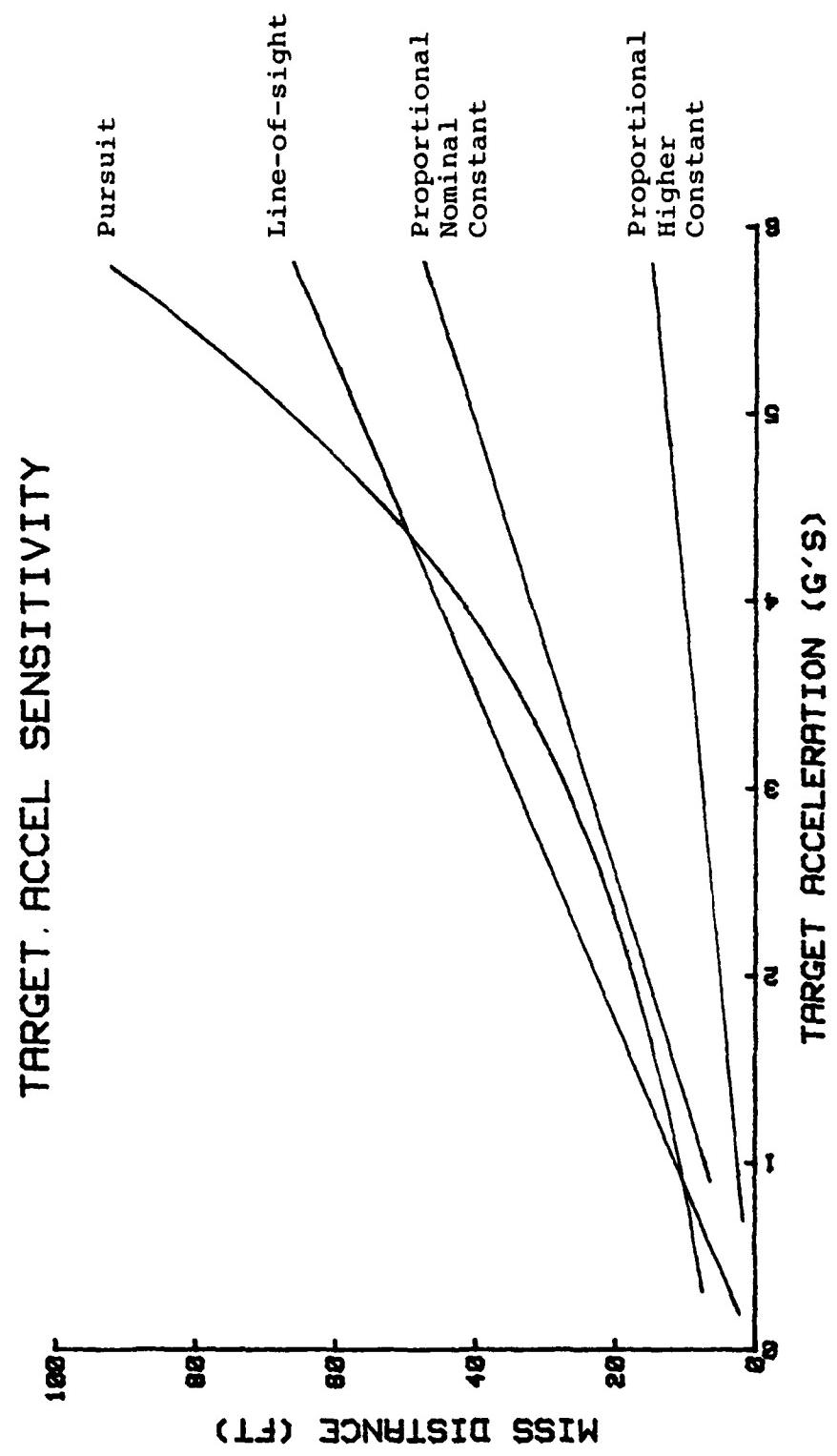


Figure (3-4)

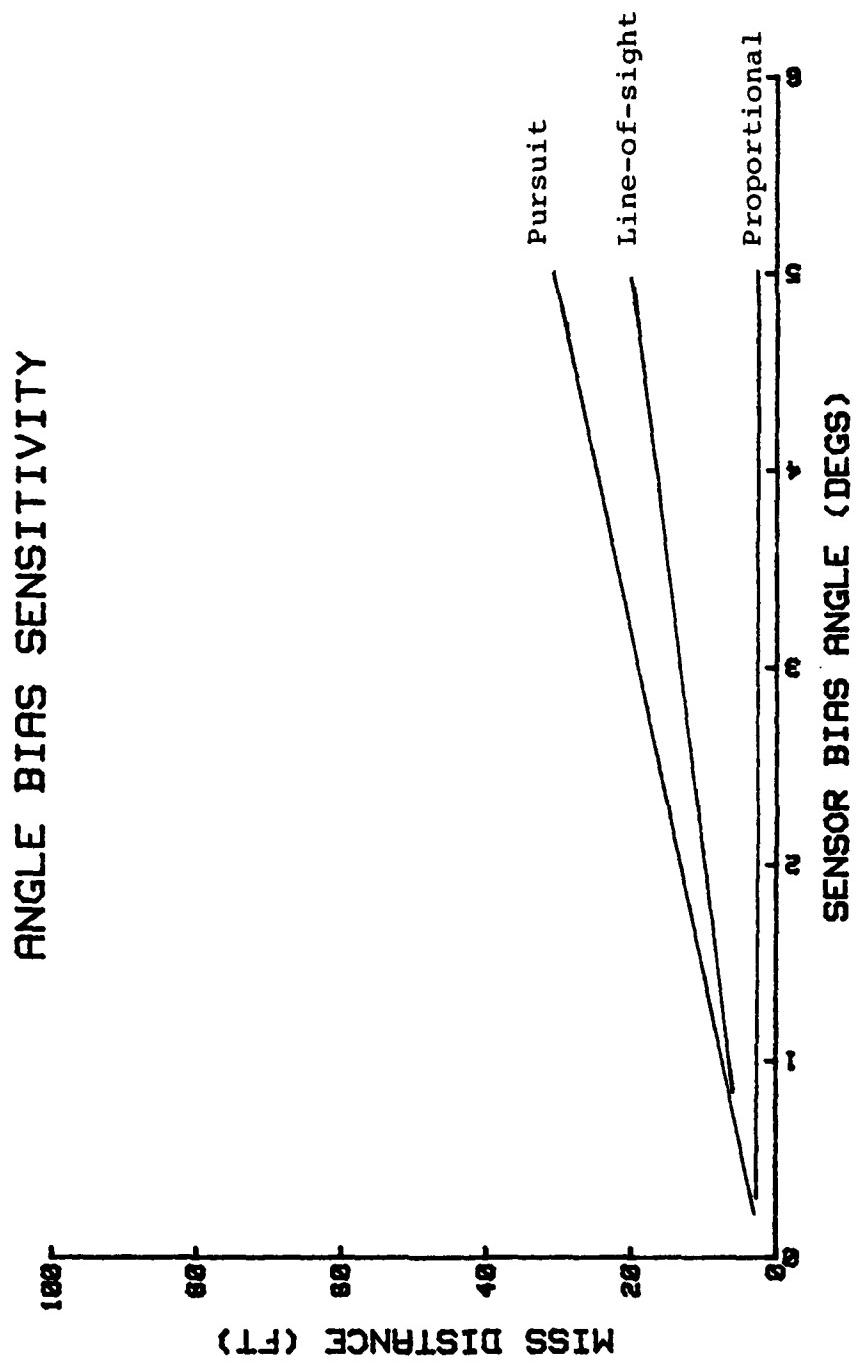


Figure (3-5)

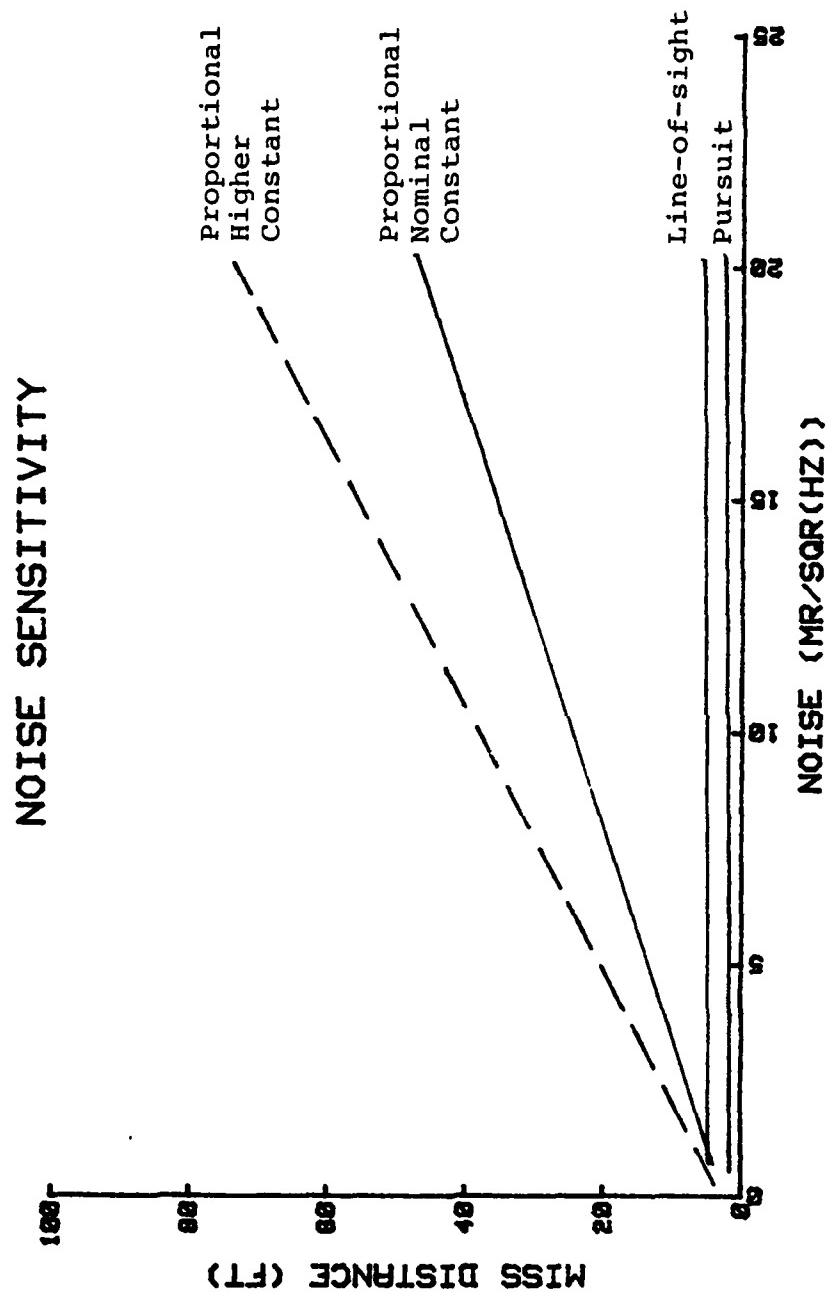


Figure (3-6)

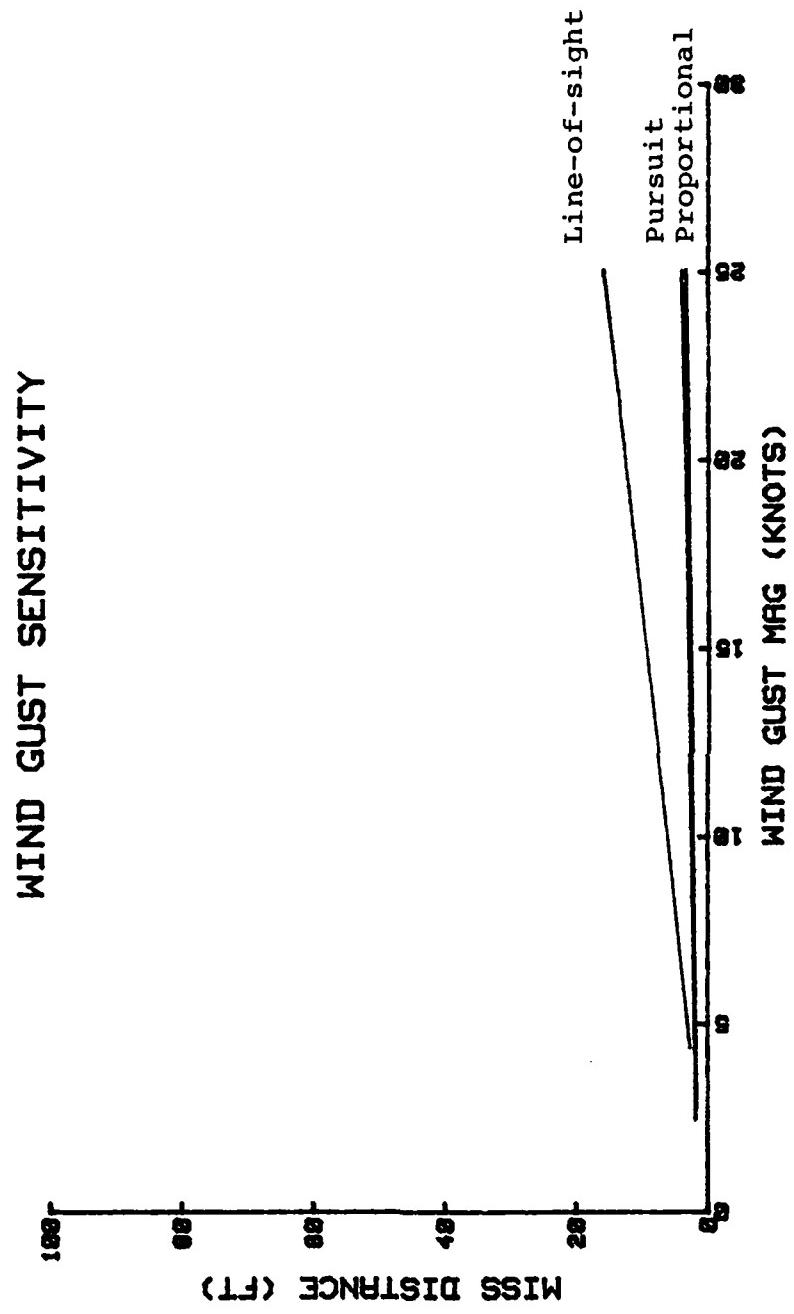


Figure (3-7)

TABLE 3-1
GUIDANCE LAW TRENDS FOR AIR TARGETS

		Target Speed	Target Acceleration	Sensor Angle Bias	Noise	Wind Gusts
Line-of-sight	Target Heading	✓	✓	✓	✓	✓
Pursuit	Good	✓	✓	✓	✓	✓
	Average	✓	✓	✓	✓	✓
	Poor	✓	✓	✓	✓	✓
Proportional	Good	✓	✓	✓	✓	✓
	Average	✓	✓	✓	✓	✓
	Poor	✓	✓	✓	✓	✓

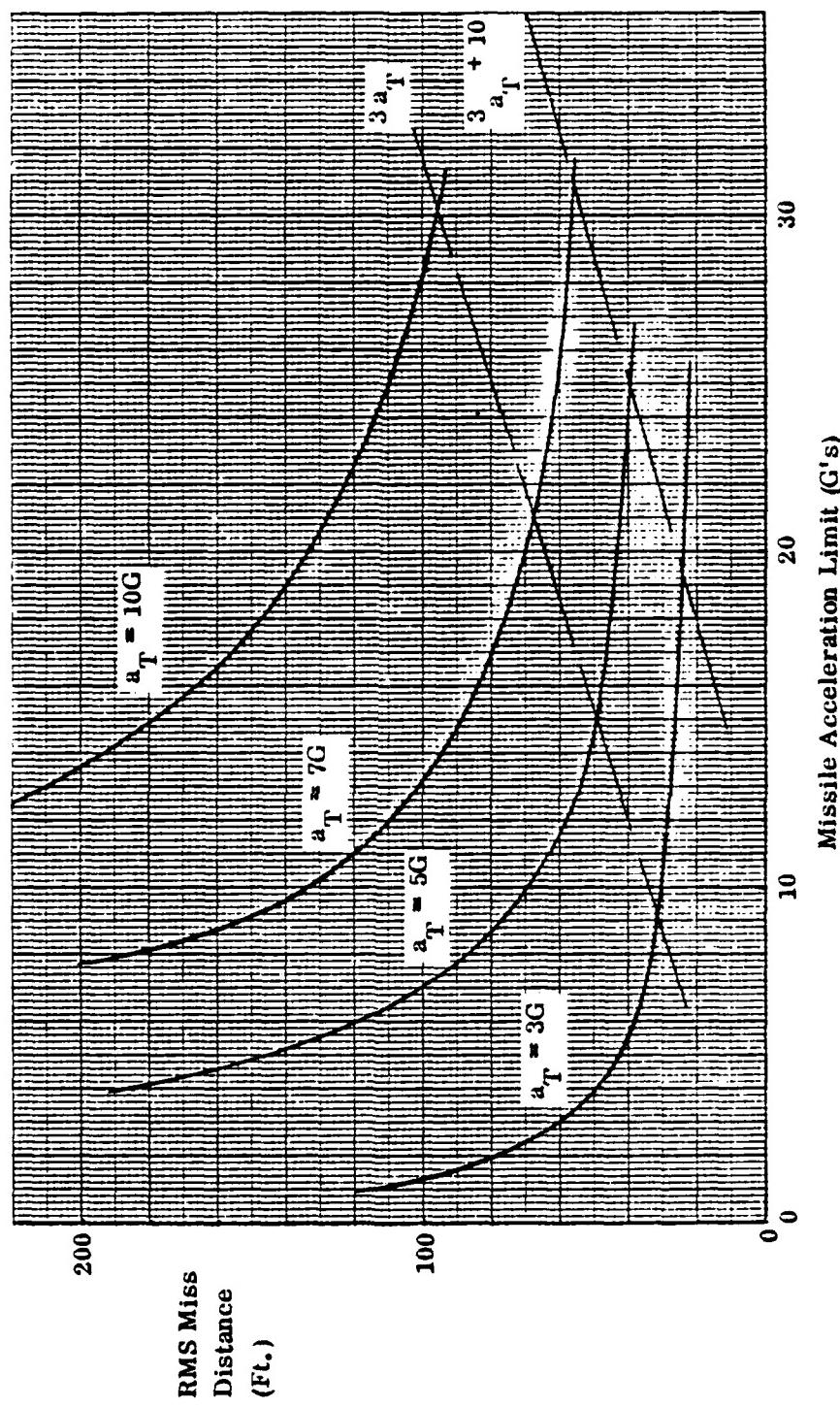


Figure (3-8). Missile acceleration limit [7].

E. PURSUIT GUIDANCE (DETAILED ANALYSIS)

As stated previously, a pursuit guidance law requires the missile velocity vector to always point at the target. For this reason the missile always ends up in a tail chase situation, with the maximum acceleration occurring at the end of the encounter. From this description the maximum acceleration of the missile can be determined.

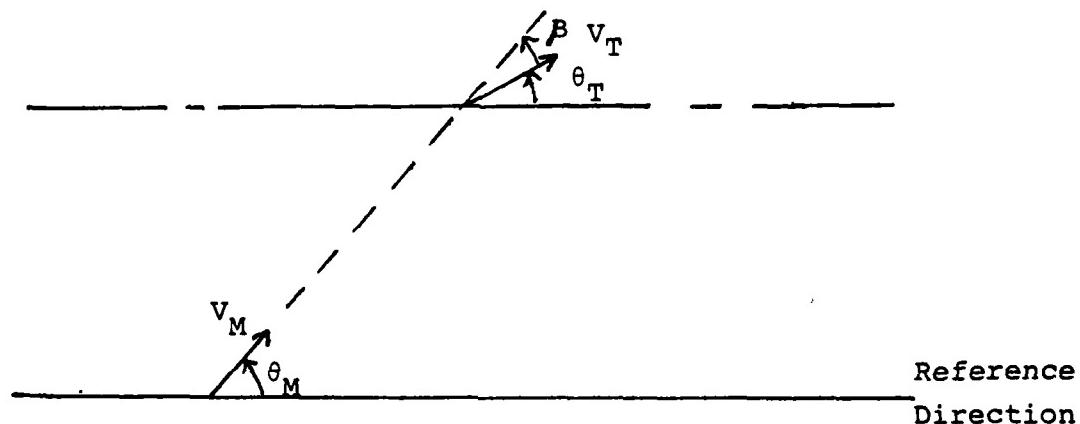


Figure (3-9). Pursuit geometry.

From Figure (3-9) the time rate of change of the range, R , is

$$\dot{R} = v_T \cos \beta - v_M$$

also

$$\dot{\beta} = -v_T \sin \beta / R$$

or

$$R \frac{d\beta}{dt} = v_T \sin \beta$$

$$\frac{d\beta}{\sin \beta} = -\frac{V_T}{R} dt \quad (1)$$

$$dt = \frac{dR}{V_T \cos \beta - V_M} \quad (2)$$

Substituting equation (2) into equation (1) gives,

$$\frac{d\beta}{\sin \beta} = -\frac{V_T}{R} \frac{dR}{(V_T \cos \beta - V_M)} ; \quad \text{Letting } k = \frac{V_M}{V_T}$$

$$\frac{(\cos \beta - k)}{\sin \beta} d\beta = -\frac{dR}{R} \quad (3)$$

Integrating equation (3) yields,

$$\ln R = k \ln(\tan \frac{\beta}{2}) - \ln(\sin \beta) + \ln c_1$$

$$\ln R = \ln \left[\frac{c_1 \tan^k \beta/2}{\sin \beta} \right]$$

By trigonometric identity,

$$(\tan \beta/2)^k = \frac{(\sin \beta)^k}{(1+\cos \beta)^k}$$

Therefore,

$$\ln R = \ln \left[\frac{c_1 (\sin \beta)^{k-1}}{(1+\cos \beta)^k} \right]$$

and

$$R = \frac{c_1 (\sin \beta)^{k-1}}{(1+\cos \beta)^k}$$

From the initial condition $\beta = \beta_0$ when $R = R_0$,

$$c_1 = \frac{R_0 (1+\cos\beta_0)^k}{(\sin\beta_0)^{k-1}}$$

$$R = R_0 \left(\frac{1+\cos\beta_0}{1+\cos\beta} \right)^k \left(\frac{\sin\beta}{\sin\beta_0} \right)^{k-1}$$

Substituting the above equation for R into the equation for $\dot{\beta}$

$$\dot{\beta} = -v_t \sin\beta/R$$

$$\dot{\beta} = -\frac{v_t}{R_0} \left(\frac{1+\cos\beta}{1+\cos\beta_0} \right)^k \frac{(\sin\beta)^{2-k}}{(\sin\beta_0)^{1-k}}$$

The missile acceleration can be expressed as a normal and a tangential component,

$$\bar{a} = v_M \dot{\beta} \hat{n} + \dot{v}_M \hat{t}$$

The normal component is a_m , where,

$$a_m = v_M \dot{\beta}$$

$$a_m = -\frac{v_M v_t}{R_0} \left(\frac{1+\cos\beta}{1+\cos\beta_0} \right)^k \frac{\sin\beta}{\sin\beta_0}^{2-k}$$

The terminal acceleration for a pursuit guidance law will occur at the end of the encounter ($\beta \rightarrow 0$). From the above expression

the terminal acceleration can be evaluated

$$a_m = 0, \quad \text{for } 1 < k < 2$$

$$a_m = -\frac{V_M V_T}{R_0} \left(\frac{2}{1+\cos\beta_0} \right)^2 \sin\beta_0 \quad \text{for } k = 2$$

$$a_m = \infty \quad \text{for } k > 2$$

Since pursuit guidance always ends up in a tail chase situation, the missile will never intercept if $k < 1$. Thus, for pursuit guidance operating against a non-maneuvering target, the velocity ratio should be between one and two. These results indicate this guidance system would not be effective against air targets; therefore, results for a maneuvering target were not pursued.

F. LINE-OF-SIGHT GUIDANCE (DETAILED ANALYSIS)

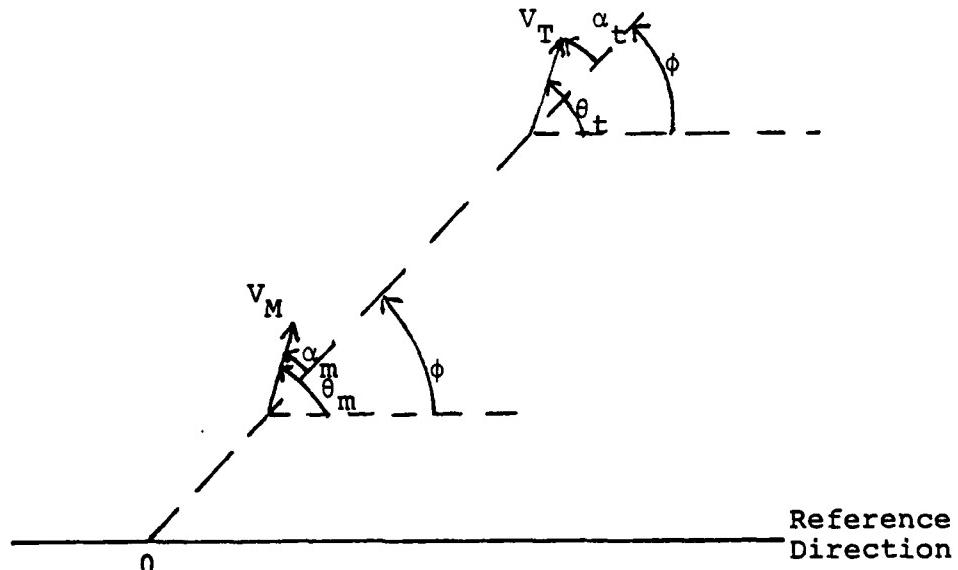


Figure (3-10). Line-of-sight geometry.

Figure (3-10) illustrates the geometry used to derive the beam rider equations of motion. The basic concept of beam rider guidance is that the missile is maintained in the line-of-sight of the target and a control point. This can be expressed in an equation as follows:

$$\dot{\phi} = \frac{v_T \sin \alpha_t}{r_t} = \frac{v_M \sin \alpha_m}{r_m} \quad (1)$$

where, r_t = range from point 0 to the target

r_m = range from point 0 to the missile

From equation (1)

$$r_t v_M \sin \alpha_m = r_m v_T \sin \alpha_t \quad (2)$$

As in the case of pursuit guidance, the missile and target accelerations can be divided into normal and tangential components. If the target is limited to constant g turns, and the normal component of missile acceleration is of interest; then,

$$\dot{v}_T = \dot{v}_M = 0$$

Differentiating equation (2) with respect to time yields,

$$\dot{r}_t v_M \sin \alpha_m + r_t v_M \dot{\alpha}_m \cos \alpha_m = \dot{r}_m v_T \sin \alpha_t + v_T \dot{\alpha}_t \cos \alpha_t$$

Solving for $\dot{\alpha}_m$,

$$\dot{\alpha}_m = \frac{1}{r_t v_M \cos \alpha_m} [r_m v_T \sin \alpha_t + r_m v_T \dot{\alpha}_t \cos \alpha_t - \dot{r}_t v_M \sin \alpha_m]$$

From the original figure,

$$\dot{r}_t = v_T \cos \alpha_t$$

$$\dot{r}_m = v_M \cos \alpha_m$$

Also

$$\theta_t = \alpha_t + \phi \longrightarrow \dot{\theta}_t = \dot{\alpha}_t + \dot{\phi}$$

$$\theta_m = \alpha_m + \phi \longrightarrow \dot{\theta}_m = \dot{\alpha}_m + \dot{\phi}$$

The target and missile accelerations (normal components) become,

$$a_t = v_T \dot{\theta}_t$$

$$a_m = v_M \dot{\theta}_m$$

Collecting equations;

$$\dot{\theta}_t = a_t/v_t$$

$$\dot{\phi} = v_T \sin \alpha_t / r_t$$

$$\dot{\alpha}_t = \dot{\theta}_t - \dot{\phi}$$

$$\dot{r}_t = v_T \cos \alpha_t$$

$$\dot{r}_m = v_M \cos \alpha_m$$

$$\begin{aligned} \dot{\alpha}_m &= \frac{1}{r_t v_M \cos \alpha_m} \left[\dot{r}_m v_T \sin \alpha_t + r_m v_T \dot{\alpha}_t \cos \alpha_t \right. \\ &\quad \left. - \dot{r}_t v_M \sin \alpha_m \right] \end{aligned}$$

$$\dot{\theta}_m = \dot{\alpha}_m + \dot{\phi}$$

$$a_m = v_M \dot{\theta}_m$$

The above equations are the equations of motion which describe the target and missile trajectories. These equations cannot be solved analytically except for highly specialized cases. The complete set of equations can be solved using a numerical integration technique. If Euler's one step method is used, the algorithm is as follows;

$$\theta_t(i+1) = \theta_t(i) + \Delta t \dot{\theta}_t(i)$$

$$\phi(i+1) = \phi(i) + \Delta t \dot{\phi}(i)$$

$$\alpha_t(i+1) = \alpha_t(i) + \Delta t \dot{\alpha}_t(i)$$

$$r_t(i+1) = r_t(i) + \Delta t \dot{r}_t(i)$$

$$r_m(i+1) = r_m(i) + \Delta t \dot{r}_m(i)$$

$$\alpha_m(i+1) = \alpha_m(i) + \Delta t \dot{\alpha}_m(i)$$

$$\theta_m(i+1) = \theta_m(i) + \Delta t \dot{\theta}_m(i)$$

With initial conditions;

$$r_t(0) = r_0$$

$$r_m(0) = 0$$

$$\phi(0) = \phi_0$$

$$\alpha_m(0) = 0$$

$$\theta_t(0) = \theta_{t_0}$$

$$\theta_m(0) = \theta_{m_0}$$

$$\alpha_t(0) = \alpha_{t_0}$$

The target and missile positions can be expressed as follows;

$$x_m(i+1) = x_m(i) + \Delta t v_M \cos \theta_m(i)$$

$$y_m(i+1) = y_m(i) + \Delta t v_M \sin \theta_m(i)$$

$$x_t(i+1) = x_t(i) + \Delta t v_T \cos \theta_t(i)$$

$$y_t(i+1) = y_t(i) + \Delta t v_T \sin \theta_t(i)$$

Where

$$x_m(0) = y_m(0) = 0$$

$$y_t(0) = r_0 \cos \phi_0$$

$$x_t(0) = r_0 \sin \phi_0$$

The above equations have been programmed on the HP 9830 computer. Table 3-II is a listing of this program. The program asks the user for the initial conditions and the target acceleration. It also asks for the integration step increment, Δt . It should be kept in mind when using the program that the error involved in integrating is of order Δt . The output is a plot of missile and target trajectories as well as the missile maximum acceleration and time of flight. Three examples follow which demonstrate possible uses of the program. (Note: All angles are input in radians.)

TABLE 3-II

```

10 PRINT "THIS PROGRAM DETERMINES THE ACCELERATION OF"
20 PRINT "A BEAM RIDER MISSILE AND PLOTS THE TRAJECTORY"
30 PRINT
40 PRINT "INPUT TIME INCREMENT FOR INTEGRATION"
50 INPUT D1
60 DIM X[250],Y[250],U[250],V[250],R[250]
70 PRINT "INPUT INITIAL TARGET RANGE"
80 INPUT R1
90 PRINT "INPUT TARGET SPEED"
100 INPUT V1
110 PRINT "INPUT MISSILE SPEED"
120 INPUT V2
130 PRINT "INPUT INITIAL LINE OF SIGHT ANGLE"
140 INPUT P1
150 PRINT "INPUT MISSILE ALPHA"
160 INPUT A2
170 PRINT "INPUT TARGET ALPHA"
180 INPUT A1
190 PRINT "INPUT MISSILE THETA"
200 INPUT T2
210 PRINT "INPUT TARGET THETA"
220 INPUT T1
230 PRINT "INPUT TARGET ACCELERATION"
240 INPUT G1
250 I=1
260 R2=0
265 R[1]=R1
270 R[1]=0
280 X[I]=0
290 Y[I]=0
300 U[I]=R1*COS(P1)
310 V[I]=R1*SIN(P1)
320 R[I]=0
330 PRINT " XM YM XT YT"
395 PRINT "DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO"
396 INPUT T9
397 IF T9=1 THEN 410
400 PRINT X[I],Y[I],U[I],V[I]
410 I=I+1
420 T3=G1/V1
430 P3=V1*SIN(A1)/R1
440 R3=T3-P3
450 R3=V1*COS(A1)
460 R4=V2*COS(A2)
461 D6=R4*V1*SIN(A1)
462 D7=R2*V1*R3*COS(A1)
463 D8=-R3*V2*SIN(A2)
464 D9=R1*V2*COS(A2)

```

TABLE 3-II (cont)

```

465 R4=(D6+D7+D8)/D9
466 T4=R4+P3
470 T1=T1+D1*T3
480 P1=P1+D1*P3
490 R1=R1+D1*R3
495 R1=R1+D1*R3
496 R2=R2+D1*R4
500 R2=R2+D1*R4
510 T2=T2+D1*T4
520 R[I]=V2*T4
530 IF ABS(R[I])<ABS(R[I-1]) THEN 600
540 M=R[I]
560 T2=R2+P1
570 R[I]=R1-R2
580 X[I]=X[I-1]+D1*V2*COS(T2)
590 Y[I]=Y[I-1]+D1*V2*SIN(T2)
640 U[I]=U[I-1]+D1*V1*COS(T1)
650 V[I]=V[I-1]+D1*V1*SIN(T1)
660 IF R[I]>0 THEN 397
665 T5=D1*(I-1)
670 PRINT "INPUT MINIMUM X VALUE"
680 INPUT X6
690 PRINT "INPUT MAXIMUM X VALUE"
700 INPUT X7
710 PRINT "INPUT MINIMUM Y VALUE"
720 INPUT Y6
730 PRINT "INPUT MAXIMUM Y VALUE"
740 INPUT Y7
800 SCALE X6,X7,Y6,Y7
810 PRINT "HAS AXIS BEEN DRAWN, 0=YES, 1=NO"
820 INPUT F1
830 IF F1=0 THEN 860
840 XAXIS 0,X7/10
850 YAXIS 0,Y7/10
860 PEN
870 FOR W=1 TO I
880 PLOT X[W],Y[W]
890 NEXT W
900 PEN
910 FOR S=1 TO I
920 PLOT U[S],V[S]
930 NEXT S
940 PEN
950 PRINT
951 PRINT
952 IF R[I]>R[I-1] THEN 954
953 M=R[I-1]
954 PRINT "THE MAXIMUM ACCELERATION IS "M" METERS/SEC/SEC"
960 PRINT "THE TIME TO INTERCEPT IS "T5" SEC"
1000 STOP

```

1. Example I (Non-maneuvering Crossing Target)

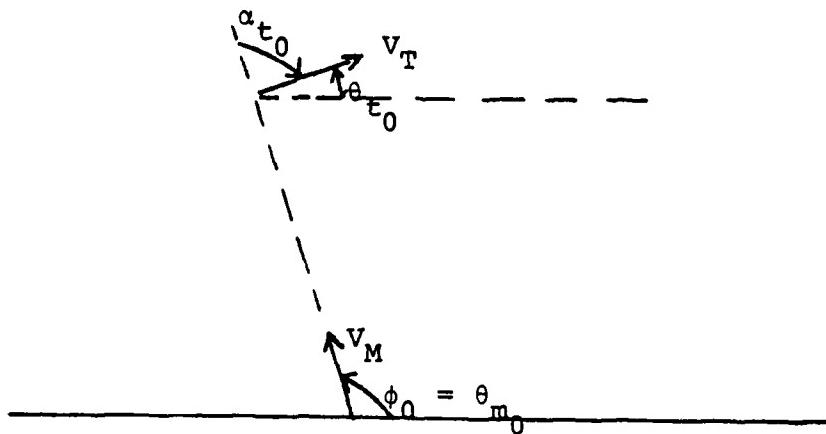


Figure (3-11). Non-maneuvering crossing geometry.

$$r_0 = 359 \text{ meters}$$

$$\phi_0 = 109.57^\circ = 1.9124 \text{ rad}$$

$$\theta_{t_0} = 17.57^\circ = .3067 \text{ rad}$$

$$\theta_{m_0} = 109.57^\circ = 1.9124 \text{ rad}$$

$$\alpha_{t_0} = -91.0^\circ = -01.5882 \text{ rad}$$

$$a_t = 0$$

$$v_M = 373 \text{ m/sec}$$

$$v_T = 221 \text{ m/sec}$$

Table 3-III is the computer output. As indicated the missile maximum acceleration is,

$$a_m = -459.25 \text{ m/sec/sec} = -46.86 \text{ g's}$$

TABLE 3-III

INPUT TIME INCREMENT FOR INTEGRATION
 INPUT INITIAL TARGET RANGE
 INPUT TARGET SPEED
 INPUT MISSILE SPEED
 INPUT INITIAL LINE OF SIGHT ANGLE
 INPUT MISSILE ALPHA
 INPUT TARGET ALPHA
 INPUT MISSILE THETA
 INPUT TARGET THETA
 INPUT TARGET ACCELERATION

XM	YM	XT	YT
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO			
0	0	-120.2644734	338.256495
-5.154984716	17.92341018	-109.7301208	341.5926479
-9.197499767	36.1300181	-99.19576818	344.9288007
-12.11631881	54.55019713	-88.66141559	348.2649534
-13.90888092	73.11385036	-78.127063	351.6011062
-14.58120543	91.75172791	-67.5927104	354.937259
-14.14756982	110.3966859	-57.05835782	358.2734118
-12.62997166	128.9848380	-46.52400523	361.6095646
-10.0574058	147.456557	-35.98965264	364.9457174
-6.464996448	165.7572973	-25.45530005	368.2818702
-1.893028052	183.8882156	-14.92094745	371.618023
3.614079320	201.6565844	-4.386594863	374.9541758
10.00880633	219.1760013	6.147757728	378.2903286
17.24154285	236.3664062	16.68211032	381.6264814
25.26153519	253.2039305	27.21646291	384.9626342
34.01774632	269.6706019	37.7508155	388.298787
43.45961099	285.7539377	48.28516809	391.6349398
53.53767559	301.4464555	58.81952068	394.9710926
64.2041194	316.745131	69.35387327	398.3072454
75.41315987	331.6508318	79.88822587	401.6433982
87.12134900	346.1677463	90.42257846	404.979551
99.28777153	360.3038325	100.956931	408.3157038
111.8741571	374.0652959	111.4912836	411.6518566
124.8449199	387.4661097	122.0256362	414.9880094
138.1671393	400.5175836	132.5599888	418.3241622
151.8104937	413.232983	143.0943414	421.660315
INPUT MINIMUM X VALUE			
INPUT MAXIMUM X VALUE			
INPUT MINIMUM Y VALUE			
INPUT MAXIMUM Y VALUE			
HAS AXIS BEEN DRAWN, 0=YES, 1=NO			

THE MAXIMUM ACCELERATION IS -459.2526072 METERS/SEC/SEC
 THE TIME TO INTERCEPT IS 1.3 SEC

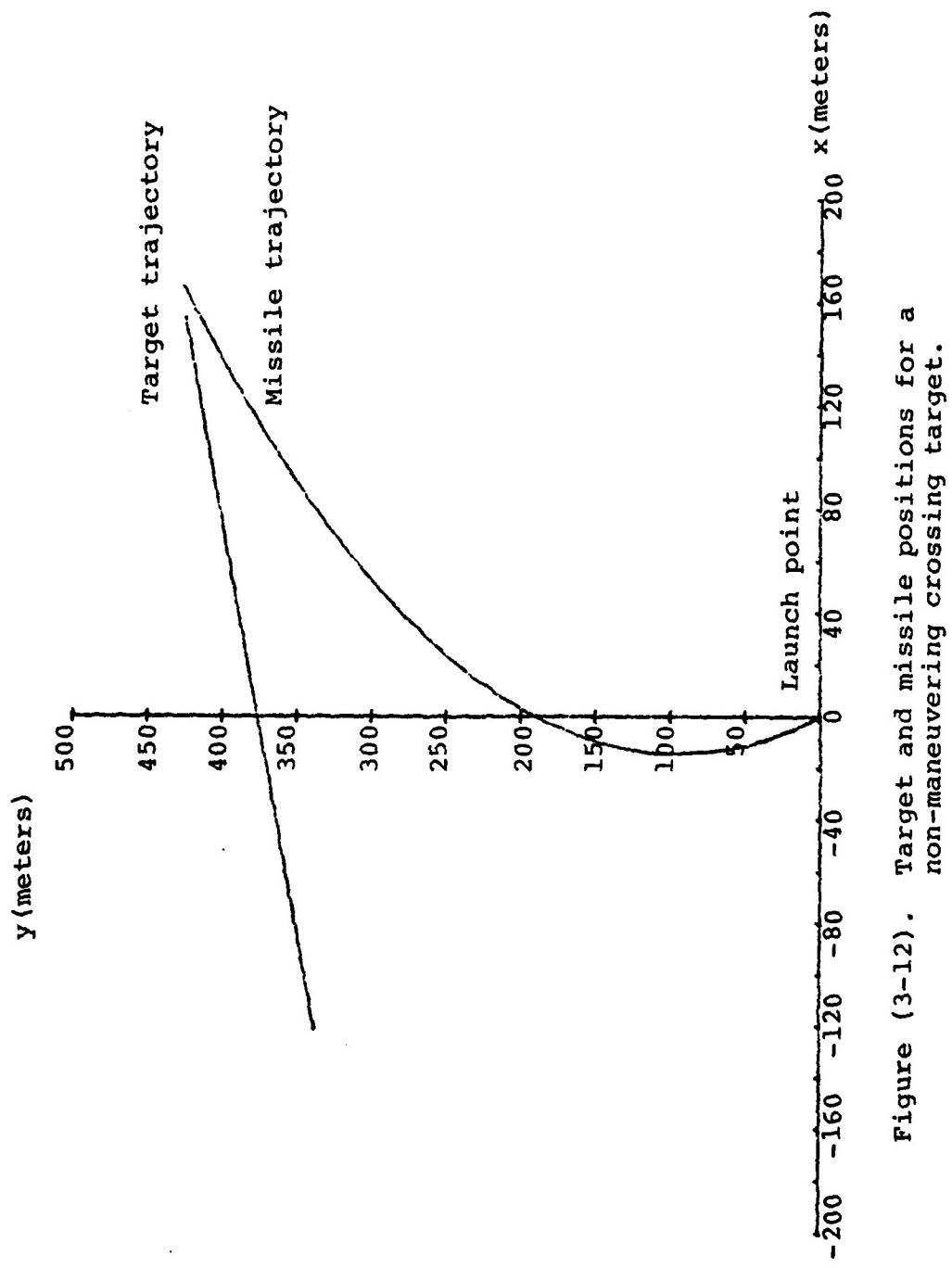


Figure (3-12). Target and missile positions for a non-maneuvering crossing target.

The large acceleration is typical of line-of-sight due to the missile requirement to stay in the beam. The trajectories are plotted in Figure (3-12).

2. Example II (Effect of V_M)

This example is presented to study the effect of missile velocity on a crossing target (non-maneuvering).

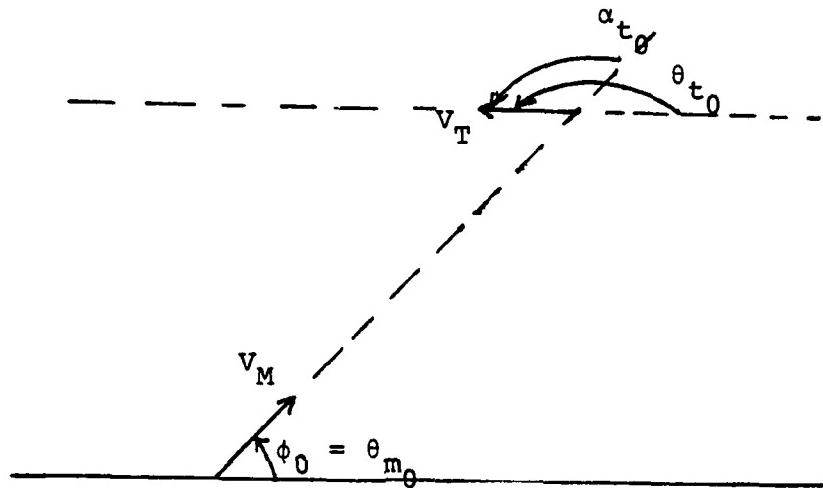


Figure (3-13). Crossing target geometry.

The initial conditions are as follows;

$$r_0 = 4000 \text{ m}$$

$$\phi_0 = 45^\circ = .7854 \text{ rad}$$

$$\theta_{m0} = 45^\circ = .7854 \text{ rad}$$

$$\alpha_{t0} = 135^\circ = 2.3562 \text{ rad}$$

$$\theta_{t0} = 180^\circ = 3.1416 \text{ rad}$$

$$v_T = 200 \text{ m/sec}$$

TABLE 3-IV

THIS PROGRAM DETERMINES THE ACCELERATION OF
A BEAM RIDER MISSILE AND PLOTS THE TRAJECTORY

INPUT TIME INCREMENT FOR INTEGRATION
INPUT INITIAL TARGET RANGE
INPUT TARGET SPEED
INPUT MISSILE SPEED = *400 M/SEC*
INPUT INITIAL LINE OF SIGHT ANGLE
INPUT MISSILE ALPHA
INPUT TARGET ALPHA
INPUT MISSILE THETA
INPUT TARGET THETA
INPUT TARGET ACCELERATION

XM YM XT YT
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

INPUT MINIMUM X VALUE

INPUT MAXIMUM X VALUE

INPUT MINIMUM Y VALUE

INPUT MAXIMUM Y VALUE

HAS AXIS BEEN DRAWN, 0=YES, 1=NO

THE MAXIMUM ACCELERATION IS 58.00129548 METERS/SEC/SEC
THE TIME TO INTERCEPT IS 8 SEC

THIS PROGRAM DETERMINES THE ACCELERATION OF
A BEAM RIDER MISSILE AND PLOTS THE TRAJECTORY

INPUT TIME INCREMENT FOR INTEGRATION
INPUT INITIAL TARGET RANGE
INPUT TARGET SPEED
INPUT MISSILE SPEED = *600 M/SEC*
INPUT INITIAL LINE OF SIGHT ANGLE
INPUT MISSILE ALPHA
INPUT TARGET ALPHA
INPUT MISSILE THETA
INPUT TARGET THETA
INPUT TARGET ACCELERATION

XM YM XT YT
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

INPUT MINIMUM X VALUE

INPUT MAXIMUM X VALUE

INPUT MINIMUM Y VALUE

INPUT MAXIMUM Y VALUE

HAS AXIS BEEN DRAWN, 0=YES, 1=NO

THE MAXIMUM ACCELERATION IS 72.78774168 METERS/SEC/SEC
THE TIME TO INTERCEPT IS 5.6 SEC

TABLE 3-IV (cont)

THIS PROGRAM DETERMINES THE ACCELERATION OF
A BEAM RIDER MISSILE AND PLOTS THE TRAJECTORY

INPUT TIME INCREMENT FOR INTEGRATION

INPUT INITIAL TARGET RANGE

INPUT TARGET SPEED

INPUT MISSILE SPEED = **800 m/sec**

INPUT INITIAL LINE OF SIGHT ANGLE

INPUT MISSILE ALPHA

INPUT TARGET ALPHA

INPUT MISSILE THETA

INPUT TARGET THETA

INPUT TARGET ACCELERATION

XM YM XT YT

DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO

INPUT MINIMUM X VALUE

INPUT MAXIMUM X VALUE

INPUT MINIMUM Y VALUE

INPUT MAXIMUM Y VALUE

HAS AXIS BEEN DRAWN, 0=YES, 1=NO

THE MAXIMUM ACCELERATION IS 87.22950871 METERS/SEC/SEC
THE TIME TO INTERCEPT IS 4.4 SEC

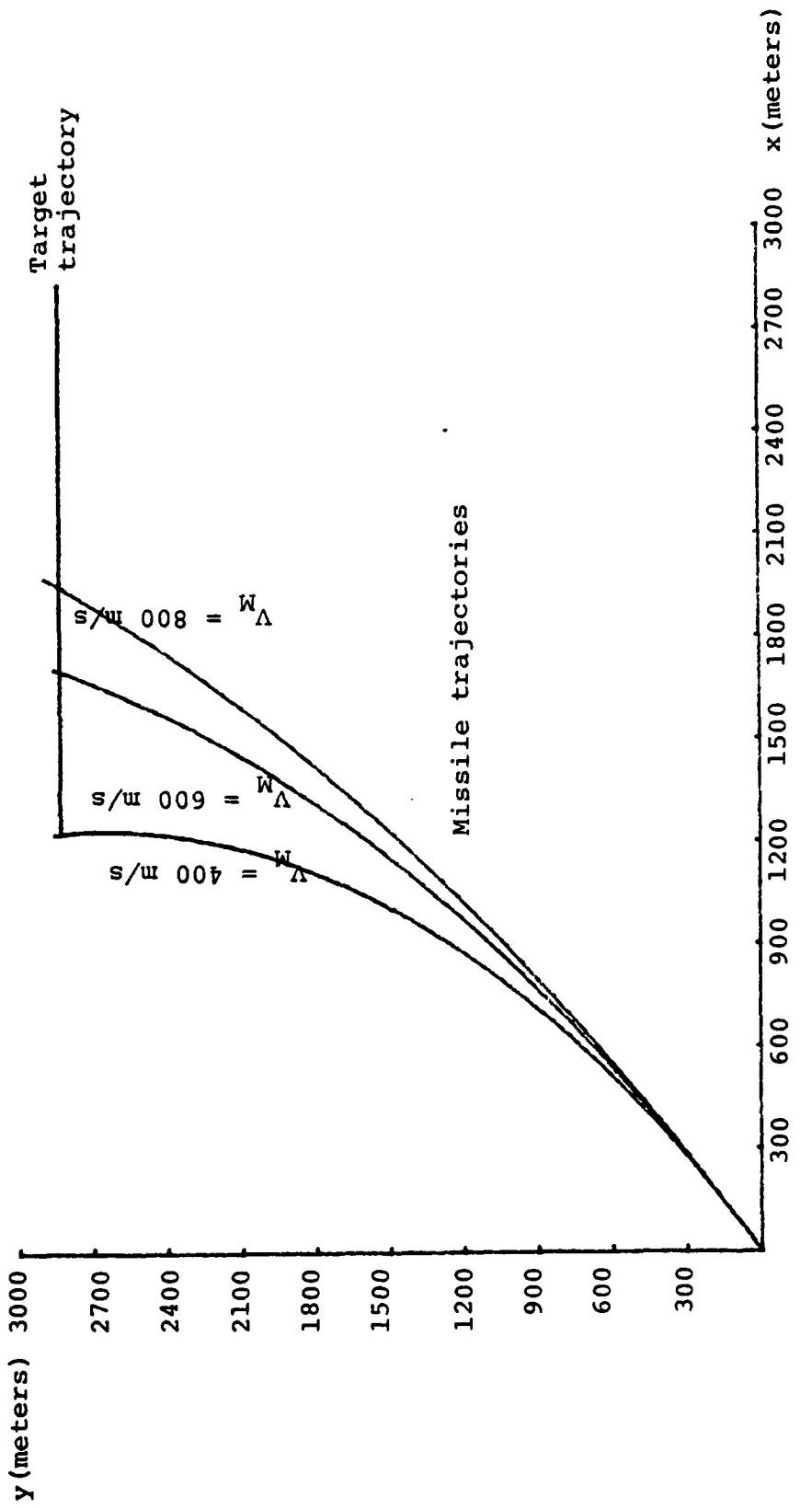


Figure (3-14). Effect of V_M on the encounter.

The program was run three times for a missile velocities of 400, 600, and 800 m/sec. Table 3-IV contains the program outputs and Figure (3-14) is the plot of the trajectories. As can be seen from the output, the missile maximum acceleration increases with increasing missile speed. For example, $V_M/V_T = 2$ gives the smallest acceleration, although the maximum acceleration for $V_M/V_T = 4$ is not exceedingly large for this scenario.

3. Example III (Maneuvering target)

In this example the effect of a target maneuver is investigated. If at the time of launch the target initiates a 7 g (68.6 m/sec/sec) turn, the following encounter would result:

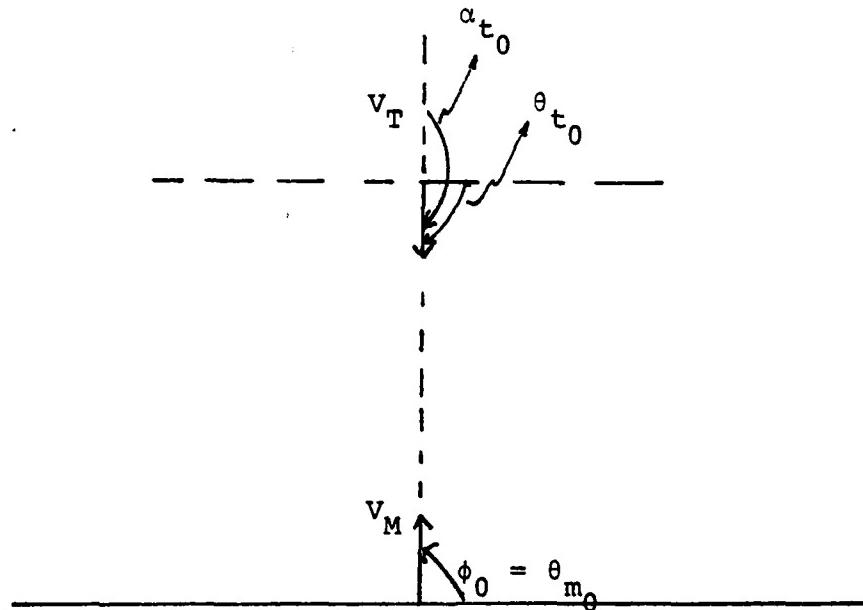


Figure (3-15). Maneuvering target.

TABLE 3-V

THIS PROGRAM DETERMINES THE ACCELERATION OF
A BEAM RIDER MISSILE AND PLOTS THE TRAJECTORY

INPUT TIME INCREMENT FOR INTEGRATION
INPUT INITIAL TARGET RANGE
INPUT TARGET SPEED
INPUT MISSILE SPEED
INPUT INITIAL LINE OF SIGHT ANGLE
INPUT MISSILE ALPHA
INPUT TARGET ALPHA
INPUT MISSILE THETA
INPUT TARGET THETA
INPUT TARGET ACCELERATION
XM YM XT YT
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO
INPUT MINIMUM X VALUE
INPUT MAXIMUM X VALUE
INPUT MINIMUM Y VALUE
INPUT MAXIMUM Y VALUE
HAS AXIS BEEN DRAWN, 0=YES, 1=NO

THE MAXIMUM ACCELERATION IS -355.8215593 METERS/SEC/SEC
THE TIME TO INTERCEPT IS 5.8 SEC

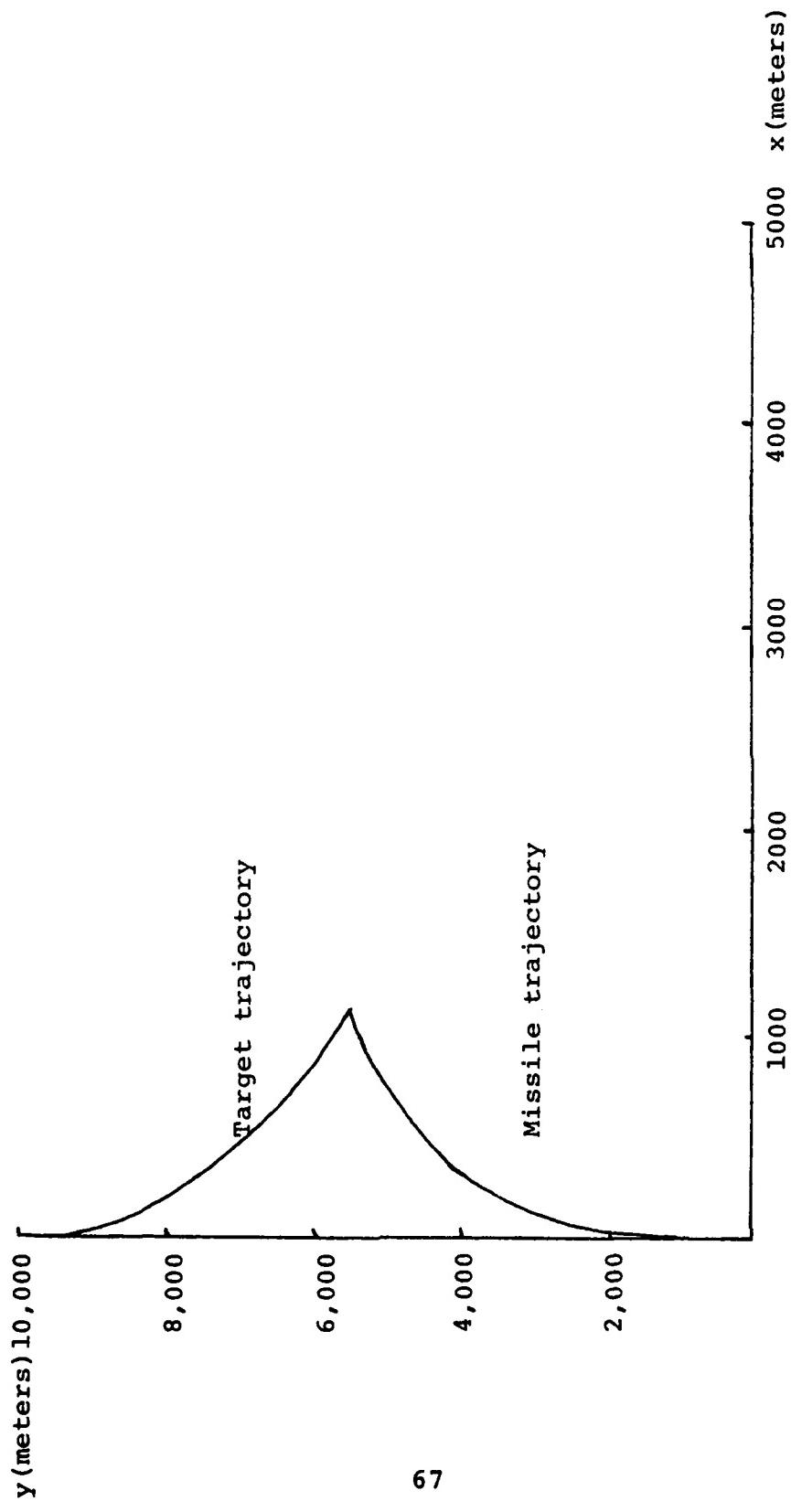


Figure (3-16). Effect of a 7g maneuver upon the encounter.

The initial conditions are;

$$r_0 = 10000 \text{ m}$$

$$v_T = 821.436 \text{ m/sec}$$

$$\phi_0 = 90^\circ = 1.5708 \text{ rad}$$

$$v_M = 985.723 \text{ m/sec}$$

$$\theta_{m0} = 90^\circ = 1.5708 \text{ rad}$$

$$\alpha_{t0} = -180^\circ$$

$$\theta_{t0} = -90^\circ$$

$$a_t = 68.60 \text{ m/sec/sec}$$

From the output (Table 3-V), notice the large missile acceleration required (211.55 m/sec/sec) to intercept a maneuvering target. The trajectories are plotted in Figure (3-16).

G. PROPORTIONAL GUIDANCE (DETAILED ANALYSIS)

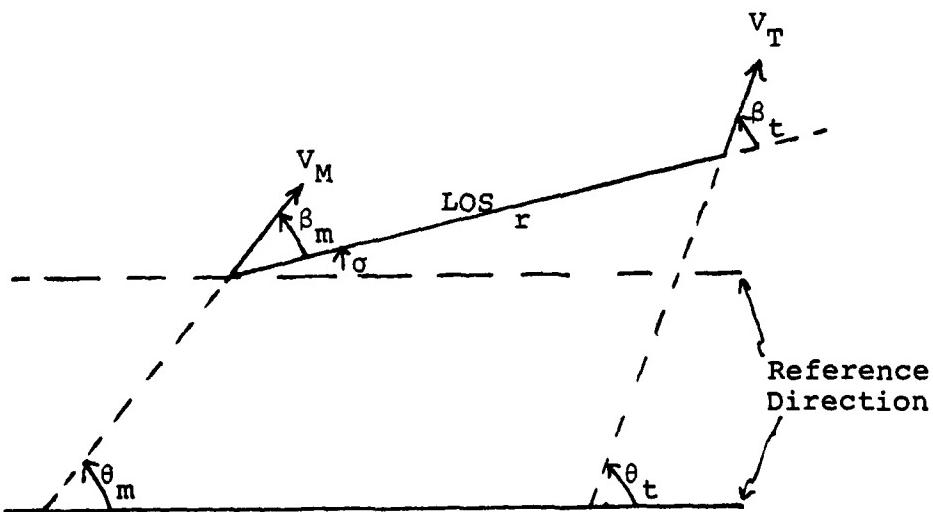


Figure (3-17). Proportional guidance geometry.

Proportional guidance automatically establishes a lead angle and reacts to a changing line-of-sight. The basic guidance law used equates the rate of change of the missile heading to a constant times the rate of change of the line-of-sight. From the above figure this law can be expressed as,

$$\dot{\theta}_m = k \dot{\sigma}$$

From the figure, the rate of change of the line-of-sight, $\dot{\sigma}$, is given by,

$$\dot{\sigma} = \frac{v_T \sin \beta_t - v_M \sin \beta_m}{r}$$

As in the case of pursuit and line-of-sight guidance, the parameter of interest here is the normal acceleration; therefore, the missile and target tangential accelerations are assumed to be zero. In this case,

$$a_m = v_M \dot{\theta}_m$$

$$a_t = v_T \dot{\theta}_t$$

From the guidance law,

$$a_m = v_M k \dot{\sigma}$$

and

$$a_m r = v_M k (v_T \sin \beta_t - v_M \sin \beta_m)$$

Since $\dot{v}_m = \dot{v}_t = 0$, the time derivative of this equation is,

$$\dot{r} \dot{a}_m + r \dot{\dot{a}}_m = k v_m [(\dot{v}_t \dot{\beta}_t \cos \beta_t - v_m \dot{\beta}_m \cos \beta_m)] \quad (1)$$

From the original figure,

$$\theta_m = \beta_m + \sigma$$

$$\dot{\beta}_m = \dot{\theta}_m - \dot{\sigma} = \dot{\theta}_m - \frac{\dot{\theta}_m}{k}$$

$$\dot{\beta}_m = \dot{\theta}_m (1 - \frac{1}{k})$$

Also,

$$\dot{\beta}_t = \dot{\theta}_t - \dot{\sigma}$$

$$\dot{\beta}_t = \dot{\theta}_t - \frac{\dot{\theta}_m}{k}$$

Making these substitutions, equation (1) becomes,

$$r \dot{\dot{a}}_m = -a_m \dot{r} + k v_M [v_T (\dot{\theta}_t - \frac{\dot{\theta}_m}{k}) \cos \beta_t - v_M \dot{\theta}_m (1 - \frac{1}{k}) \cos \beta_m]$$

Since,

$$a_m = v_M \dot{\theta}_m$$

$$a_t = v_T \dot{\theta}_t$$

$$r \dot{\dot{a}}_m = -a_m \dot{r} + k v_M a_t \cos \beta_t - k v_M a_m \cos \beta_m$$

$$- a_m (v_T \cos \beta_t - v_M \cos \beta_m)$$

From Figure (3-17),

$$\dot{r} = V_T \cos \beta_t - V_M \cos \beta_m$$

$$r \dot{a}_m = -2 a_m \dot{r} + k V_M a_t \cos \beta_t - k V_M a_m \cos \beta_m$$

$$\dot{a}_m = \frac{k V_M a_t \cos \beta_t}{r} - \frac{a_m}{r} [2\dot{r} + k V_M \cos \beta_m]$$

Collecting equations;

$$\dot{r} = V_T \cos \beta_t - V_M \cos \beta_m$$

$$\dot{\beta}_t = \dot{\theta}_t - \dot{\sigma}$$

$$\dot{\beta}_m = \dot{\theta}_m - \dot{\sigma}$$

$$\dot{\sigma} = \frac{V_T \sin \beta_t - V_M \sin \beta_m}{r}$$

$$\dot{\theta}_t = \frac{a_t}{V_T}$$

$$\dot{\theta}_m = k \dot{\sigma}$$

The above equations are the equations of motion for a missile using proportional navigation assuming constant missile and target speeds. As with the line-of-sight equations, the motion is quite complex. The equations cannot be solved analytically except for special cases. One such case will be investigated here. That is for a non-maneuvering crossing target.

1. Example IV (Non-Maneuvering Crossing Target)

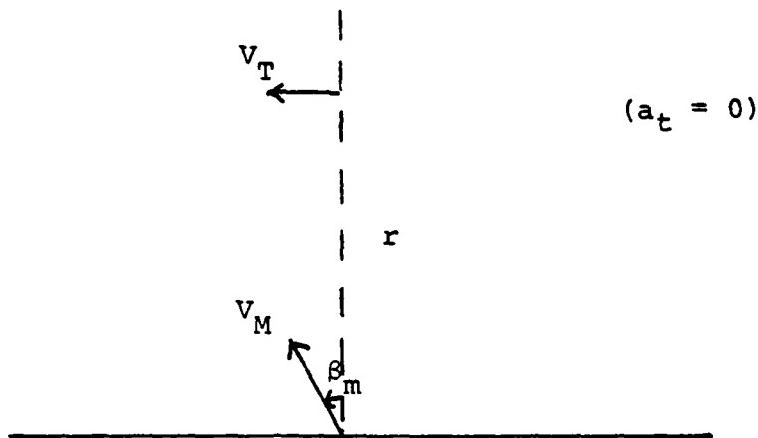


Figure (3-18). Non-maneuvering target.

For this case, $\dot{r} = -v_M \cos \beta_m$ from equation (1)

$$\dot{a}_m = -\frac{a_m}{r} [2\dot{r} - kr']$$

$$\frac{\dot{a}_m}{a_m} = \frac{\dot{r}}{r} (k-2)$$

$$\ln a_m = (k-2) \ln r + \ln c_1$$

If

$$a_m = a_0 \text{ at } r = r_0$$

$$a_m = a_0 \left(\frac{r}{r_0}\right)^{k-2}$$

From this equation,

$$\text{if } k > 2 \quad a_m \rightarrow 0 \text{ as } r \rightarrow 0$$

if $k = 2$ $a_m = a_0 = \text{constant}$

if $k < 2$ $a_m \rightarrow \infty$ as $r \rightarrow 0$

From the above example it can be realized that the proportionality constant, k , must be greater than two. A more general analysis of the equations of motion can be obtained by solving the equations numerically. The same Euler's one step method is used here with initial conditions, at $t = 0$,

$$a_m(0) = a_{m0}$$

$$r(0) = r_0$$

$$\beta_t(0) = \beta_{t0}$$

$$\beta_m(0) = \beta_{m0}$$

$$\sigma(0) = \sigma_0$$

$$\theta_t(0) = \theta_{t0}$$

$$\theta_m(0) = \theta_{m0}$$

The algorithm used is as follows:

$$r(i+1) = r(i) + \Delta t \dot{r}(i)$$

$$a_m(i+1) = a_m(i) + \Delta t \dot{a}_m(i)$$

$$\sigma(i+1) = \sigma(i) + \Delta t \dot{\sigma}(i)$$

$$\theta_m(i+1) = \theta_m(i) + \Delta t \dot{\theta}_m(i)$$

$$\theta_t^{(i+1)} = \theta_t^{(i)} + \Delta t \dot{\theta}_t^{(i)}$$

$$\beta_m^{(i+1)} = \beta_m^{(i)} + \Delta t \dot{\beta}_m^{(i)}$$

$$\beta_t^{(i+1)} = \beta_t^{(i)} + \Delta t \dot{\beta}_t^{(i)}$$

The trajectory of the missile and target can be determined assuming the missile is at the origin at $t = 0$. From the original figure the missile and target positions are given by,

$$x_m^{(i+1)} = x_m^{(i)} + \Delta t v_M \cos \theta_m^{(i)}$$

$$y_m^{(i+1)} = y_m^{(i)} + \Delta t v_M \sin \theta_m^{(i)}$$

$$x_t^{(i+1)} = x_t^{(i)} + \Delta t v_T \cos \theta_t^{(i)}$$

$$y_t^{(i+1)} = y_t^{(i)} + \Delta t v_T \sin \theta_t^{(i)}$$

With initial conditions,

$$x_m(0) = y_m(0) = 0$$

$$x_t(0) = r_0 \cos \sigma_0$$

$$y_t(0) = r_0 \sin \sigma_0$$

Evaluating the missile initial acceleration can be more complicated. One procedure which is both realistic and of interest is to have the missile and target on a constant bearing - decreasing range course ($a_t = 0$) when the target initiates a constant g turn at $t = 0$. In this case $a_m(0) = 0$ and the subsequent motion can be found.

TABLE 3-VI

```

10 PRINT "THIS PROGRAM FINDS THE MAXIMUM"
20 PRINT "MISSILE ACCELERATION FOR A"
30 PRINT "PROPORTIONAL NAVIGATION SYSTEM"
35 PRINT
36 DIM X[250],Y[250],U[250],V[250]
38 DIM R[250],T[250],R[250]
40 PRINT "INPUT TIME INCREMENT"
50 INPUT D1
60 PRINT "INPUT NAVIGATION CONSTANT"
70 INPUT K
80 PRINT "INPUT MISSILE VELOCITY"
90 INPUT V1
100 PRINT "INPUT TARGET VELOCITY"
110 INPUT V2
120 PRINT "INPUT TARGET ACCELERATION"
130 INPUT A1
140 PRINT "INPUT INITIAL MISSILE ACCELERATION"
150 INPUT A[1]
160 PRINT "INPUT INITIAL RANGE"
170 INPUT R[1]
180 PRINT "INPUT BETA TARGET"
190 INPUT B1
210 PRINT "INPUT BETA MISSILE"
220 INPUT B2
230 PRINT "INPUT THETA TARGET"
240 INPUT T1
250 PRINT "INPUT THETA MISSILE"
260 INPUT T2
270 PRINT "INPUT SIGMA"
280 INPUT S1
290 I=1
295 PRINT "      MISSILE POSIT          TARGET POSIT"
296 PRINT
297 PRINT "      X1           Y1           X2           Y2"
300 X[I]=0
310 Y[I]=0
320 U[I]=R[1]*COS(S1)
330 V[I]=R[1]*SIN(S1)
340 PRINT "DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO"
345 INPUT Q1
350 IF Q1=1 THEN 365
360 PRINT X[I],Y[I],U[I],V[I]
365 I=I+1
370 X[I]=X[I-1]+V1*COS(T2)*D1
380 Y[I]=Y[I-1]+V1*D1*SIN(T2)
390 U[I]=U[I-1]+V2*D1*COS(T1)
400 V[I]=V[I-1]+D1*V2*SIN(T1)
410 R2=V2*COS(B1)-V1*COS(B2)
420 R8=K*V1*A1*COS(B1)/R[I-1]

```

TABLE 3-VI (cont)

```

430 R9=R[I-1]*(2*R2+K*V1*COS(B2))/R[I-1]
440 R3=R8-R9
445 T[I]=(I-1)*D1
450 S2=(V2*SIN(B1)-V1*SIN(B2))/R[I-1]
460 T8=R1/V2
470 T9=K*S2
480 B3=T8-S2
490 B4=T9-S2
500 R[I]=R[I-1]+D1*R2
510 A[I]=A[I-1]+D1*R3
520 S1=S1+D1*S2
530 T1=T1+D1*T8
540 T2=T2+D1*T9
550 B1=B1+D1*B3
560 B2=B2+D1*B4
570 IF A[I]<A[I-1] THEN 600
580 Z1=A[I]
590 IF R[I]>0 THEN 350
601 IF ABS(A[I])<ABS(A[I-1]) THEN 603
602 Z1=A[I-1]
603 Z2=T[I]
618 PRINT
619 PRINT
620 PRINT "MAXIMUM MISSILE ACC IS"Z1" METERS/SEC/SEC"
640 PRINT "MISSILE TIME OF FLIGHT IS" "Z2" SEC"
700 PRINT "INPUT MINIMUM VALUE OF X"
710 INPUT X6
720 PRINT "INPUT MAXIMUM VALUE OF X"
730 INPUT X7
740 PRINT "INPUT MINIMUM VALUE OF Y"
750 INPUT Y6
760 PRINT "INPUT MAXIMUM VALUE OF Y"
770 INPUT Y7
780 SCALE X6,X7,Y6,Y7
790 PRINT "HAVE THE AXIS BEEN DRAWN, 0=YES,1=NO"
791 INPUT F1
792 IF F1=0 THEN 830
800 XAXIS 0,X7/10
810 YAXIS 0,Y7/10
830 PEN
840 FOR J=1 TO I
850 PLOT X[J],Y[J]
860 NEXT J
870 PEN
880 FOR W=1 TO I
890 PLOT U[W],V[W]
900 NEXT W
910 PEN
1000 STOP

```

The above algorithm was programmed on the HP 9830 computer. The listing is included in Table 3-VI. The inputs and output of the program are the same as the line-of-sight guidance program. Several examples follow which demonstrates the use of the program.

2. Example V (Crossing Maneuvering Target)

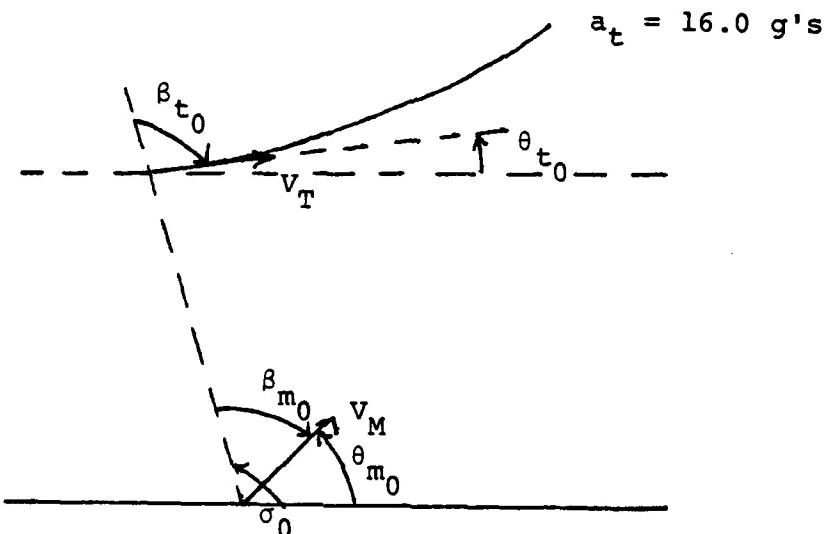


Figure (3-19). Crossing maneuvering target.

$$v_M = 373 \text{ m/sec}$$

$$\theta_{t0} = 17.57^\circ = 0.3067 \text{ rad}$$

$$v_T = 221 \text{ m/sec}$$

$$\beta_{t0} = -91.0^\circ = -1.5882 \text{ rad}$$

$$a_t = 156.8 \text{ m/sec/sec}$$

$$\beta_{m0} = -38.0^\circ = -0.6632 \text{ rad}$$

$$r_0 = 359 \text{ m}$$

$$\sigma_0 = 109.57^\circ = 1.9124 \text{ rad}$$

$$\theta_{m0} = 71.57^\circ = 1.2491 \text{ rad}$$

The output is listed in Table 3-VII. An important aspect of this problem is the maximum acceleration required (202.92 m/sec/sec).

TABLE 3-VII

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

```
INPUT TIME INCREMENT
INPUT NAVIGATION CONSTANT
INPUT MISSILE VELOCITY
INPUT TARGET VELOCITY
INPUT TARGET ACCELERATION
INPUT INITIAL MISSILE ACCELERATION
INPUT INITIAL RANGE
INPUT BETA TARGET
INPUT BETA MISSILE
INPUT THETA TARGET
INPUT THETA MISSILE
INPUT SIGMA
MISSILE POSIT
```

MISSILE POSIT TARGET POSIT

X1	Y1	X2	Y2
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO			
0	0	-120.2644734	338.256495
5.896688382	17.69326330	-109.7301208	341.5926479
11.72925942	35.40776626	-99.32072172	345.3003306
17.50036490	53.14238894	-89.04937488	349.3748777
23.21064114	70.89669110	-78.92900524	353.8111620
28.85860473	88.67031421	-68.97234776	358.603601
34.44056825	106.4659749	-59.19193143	363.7461642
39.95057982	124.2834458	-49.60006345	369.2323805
45.38038997	142.1255214	-40.20881377	375.0553462
50.71944940	159.994963	-31.02999988	381.2077340
55.95494106	177.8950225	-22.07517195	387.6818026
61.07184968	195.82934	-13.3555983	394.4694036
66.0530711	213.8018172	-4.882251242	401.5619976
70.87956373	231.8164631	3.334206794	408.9506590
75.53054295	249.8772187	11.2834366	416.6260903
79.98371892	267.9877605	18.95543527	424.578633
84.21557618	286.1512922	26.34054872	432.7982802
88.20189238	304.370331	33.42948387	441.2746885
91.91709127	322.6464989	40.21332036	449.9971918
95.33662288	340.9803284	46.68352174	458.9548139
98.43536132	359.3710956	52.8319462	468.136283
101.1890086	377.8166893	58.65085689	477.5300458
103.5742899	396.3135251	64.13293155	487.1242815
105.5693244	414.8565113	69.27127182	496.906917
107.1539546	433.4390689	74.05941184	506.8656426
108.3100167	452.0532039	78.49132646	516.9879266
109.0215346	470.6896263	82.56143878	527.2610316
109.2748269	489.3379062	86.26462715	537.6720304
109.0585142	507.9866517	89.59623167	548.2078223
108.3634271	526.6236943	92.55206	558.8551496
107.1824200	545.2362632	95.12839268	569.6006142
105.5101054	563.811135	97.32198776	580.4306946
103.3425418	582.3347461	99.13008494	591.3317626
100.6769169	600.7932657	100.550409	602.2991010

TABLE 3-VII (cont)

MAXIMUM MISSILE ACC IS 202.9212389 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 1.7 SEC
INPUT MINIMUM VALUE OF X
INPUT MAXIMUM VALUE OF X
INPUT MINIMUM VALUE OF Y
INPUT MAXIMUM VALUE OF Y
HAVE THE AXIS BEEN DRAWN, 0=YES,1=NO

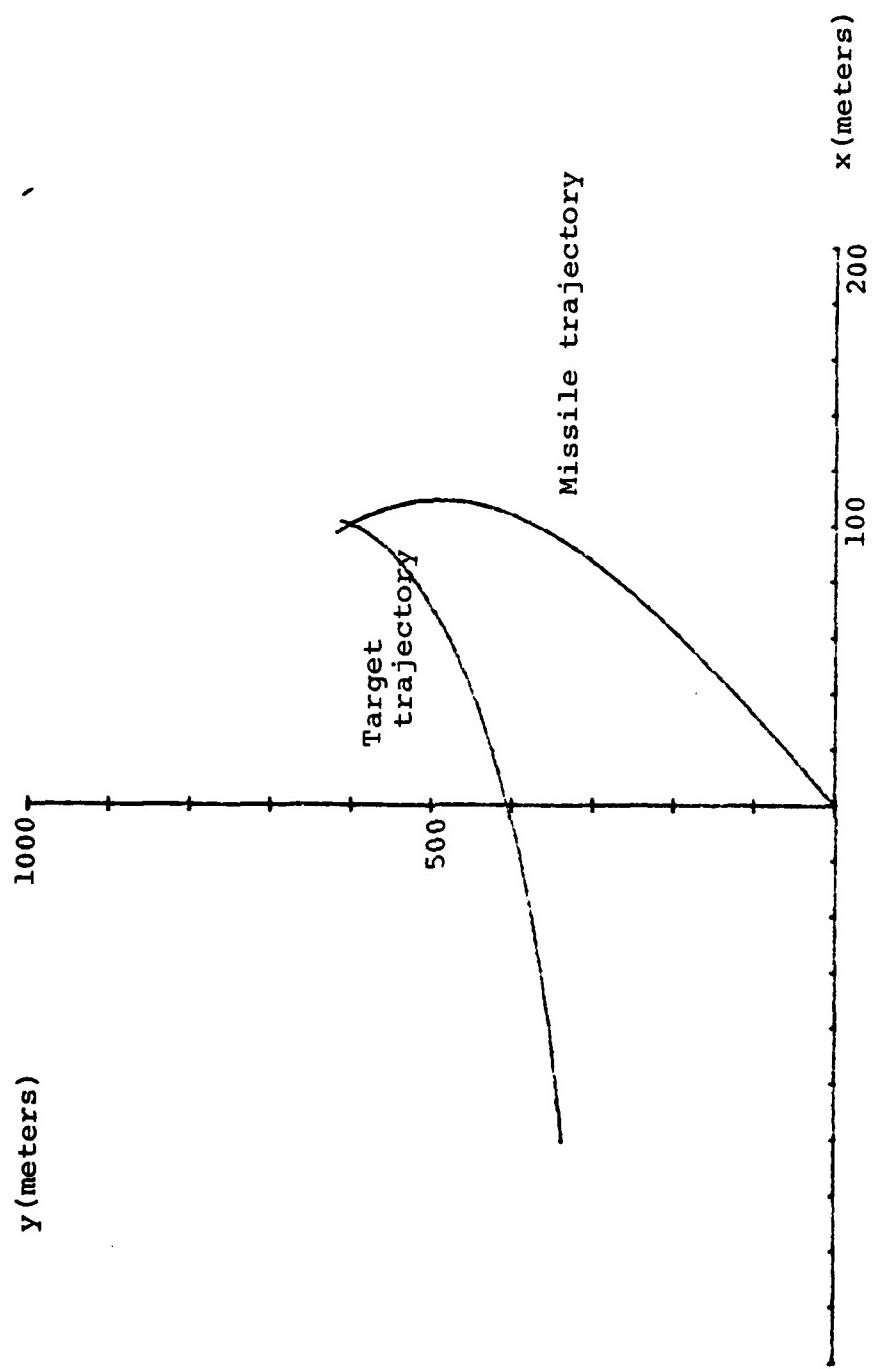


Figure (3-20). Trajectories of crossing, maneuvering intercept.

The same problem was run for line-of-sight guidance with no target acceleration in Example I. The maximum acceleration was 459.25 m/sec/sec. This points out the advantage of proportional navigation over line-of-sight guidance. Figure (3-20) is a plot of the trajectories.

3. Example VI (Effect of k)

This example demonstrates the effect of varying the proportionality constant, k . The scenario is as follows.

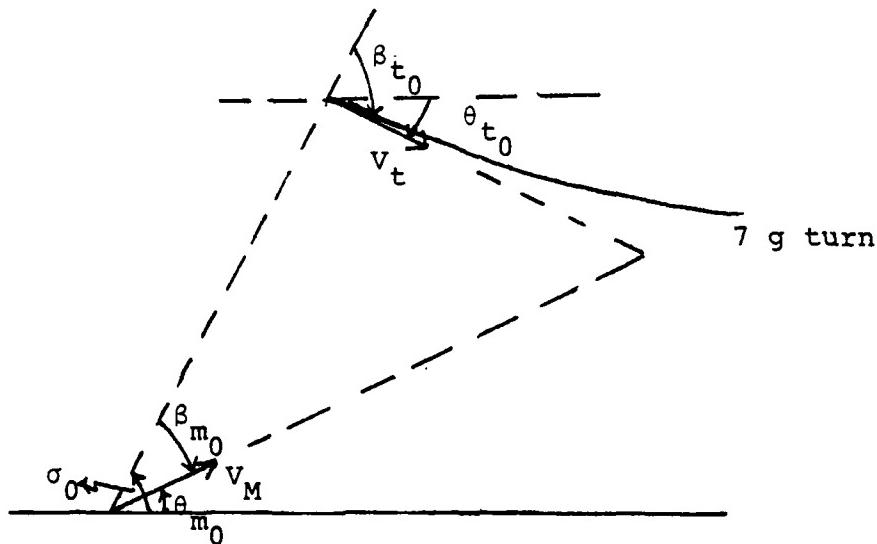


Figure (3-21). Initial geometry.

$$v_t = 208 \text{ m/sec}$$

$$v_m = 413 \text{ m/sec}$$

$$a_t = 68.60 \text{ m/sec/sec}$$

$$\beta_{t0} = -106^\circ = -1.85 \text{ rad}$$

$$\beta_{m0} = -29^\circ = -.5061 \text{ rad}$$

TABLE 3-VIII

```
INPUT TIME INCREMENT  
INPUT NAVIGATION CONSTANT = 3  
INPUT MISSILE VELOCITY  
INPUT TARGET VELOCITY  
INPUT TARGET ACCELERATION  
INPUT INITIAL MISSILE ACCELERATION  
INPUT INITIAL RANGE  
INPUT BETA TARGET  
INPUT BETA MISSILE  
INPUT THETA TARGET  
INPUT THETA MISSILE  
INPUT SIGMA  
MISSILE POSIT TARGET POSIT
```

X1 Y1 X2 Y2
DO YOU WANT A PRINT OF THE OUTPUT: 0=YES, 1=NO

```
MAXIMUM MISSILE ACC IS 83.70521471    METERS/SEC/SEC  
MISSILE TIME OF FLIGHT IS     3.2    SEC  
INPUT MINIMUM VALUE OF X  
INPUT MAXIMUM VALUE OF X  
INPUT MINIMUM VALUE OF Y  
INPUT MAXIMUM VALUE OF Y  
HAVE THE AXIS BEEN DRAWN, 0=YES, 1=NO
```

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

```
INPUT TIME INCREMENT  
INPUT NAVIGATION CONSTANT = 4  
INPUT MISSILE VELOCITY  
INPUT TARGET VELOCITY  
INPUT TARGET ACCELERATION  
INPUT INITIAL MISSILE ACCELERATION  
INPUT INITIAL RANGE  
INPUT BETA TARGET  
INPUT BETA MISSILE  
INPUT THETA TARGET  
INPUT THETA MISSILE  
INPUT SIGMA  
MISSILE POSIT
```

X1 Y1 X2 Y2
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

MAXIMUM MISSILE ACC IS 71.11333197 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 8.2 SEC

TABLE 3-VIII (cont)

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

INPUT TIME INCREMENT
INPUT NAVIGATION CONSTANT = 5
INPUT MISSILE VELOCITY
INPUT TARGET VELOCITY
INPUT TARGET ACCELERATION
INPUT INITIAL MISSILE ACCELERATION
INPUT INITIAL RANGE
INPUT BETA TARGET
INPUT BETA MISSILE
INPUT THETA TARGET
INPUT THETA MISSILE
INPUT SIGMA
MISSILE POSIT TARGET POSIT

X1 Y1 X2 Y2
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

MAXIMUM MISSILE RCC IS 65.3857248 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 3.2 SEC
INPUT MINIMUM VALUE OF X
INPUT MAXIMUM VALUE OF X
INPUT MINIMUM VALUE OF Y
INPUT MAXIMUM VALUE OF Y
HAVE THE AXIS BEEN DRAWN, 0=YES, 1=NO

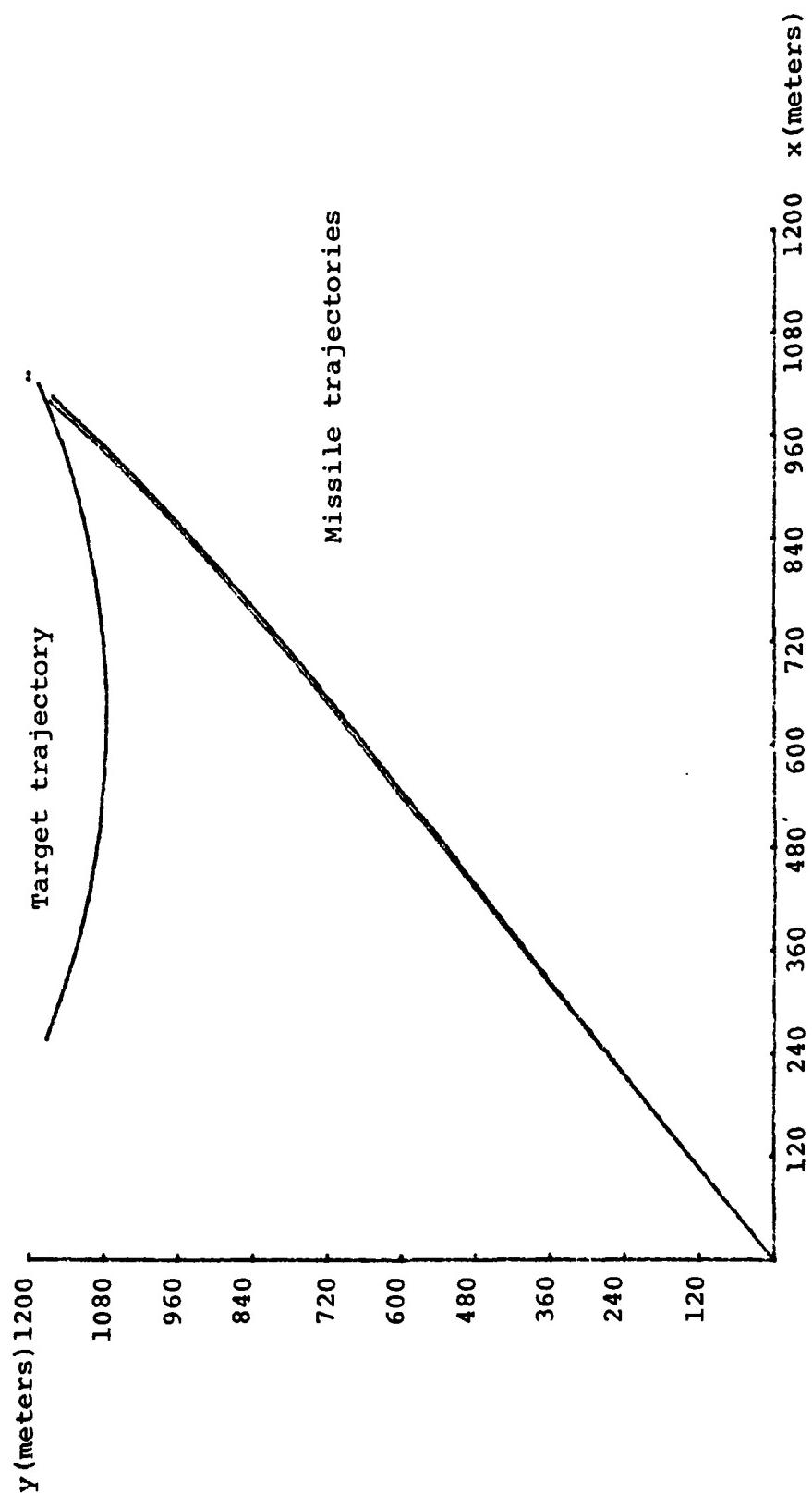


Figure (3-22). Effect of varying k .

$$\theta_{m_0} = 48.6^\circ = .8483 \text{ rad}$$

$$\theta_{t_0} = -28.4^\circ = -.4956 \text{ rad}$$

$$\sigma_0 = 77.6^\circ = 1.3544 \text{ rad}$$

The problem was run for $k = 3, 4$, and 5 .

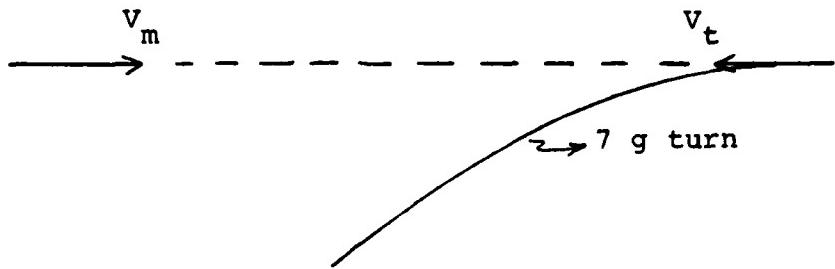
Table 3-VIII is the output. It can be seen that the effect of increasing the proportionality constant is to decrease the maximum acceleration required. Figure (3-22) is a plot of the trajectories.

H. DESIGN EXAMPLE (GUIDANCE LAW SELECTION)

From the examples given in this chapter, it can be seen that a missile designed to encounter a highly maneuverable target, such as a fighter, requires a proportional guidance law to limit the maximum acceleration required of the missile. To select the proportionality constant, it was assumed that the threat could maintain a constant $7g$ turn at $M_t = 1.5$ and an altitude of 10,000 feet.

Three cases were investigated, (1) A head-on encounter with the target initiating a turn at 10,000 meters range, (2) A crossing encounter in which the target turns into the missile at 10,000 meters range, and (3) An oblique, closing encounter in which the target turns into the missile. The scenario and computer outputs are shown in Figures (3-23), (3-24), and (3-25). From this analysis the crossing encounter requires the largest acceleration (263.2 m/sec/sec).

For the crossing scenario then, the missile speed was varied and the results indicated that as the speed increased the maximum



$$\begin{aligned}
 v_m &= 821 \text{ m/sec } (M = 2.5) & \beta_{m_0} &= 0 \\
 v_t &= 492 \text{ m/sec } (M = 1.5) & \beta_{t_0} &= 180^\circ \\
 r_0 &= 10,000 \text{ m} & \sigma_0 &= 0 \\
 \theta_{m_0} &= 0 & k &= 4 \\
 \theta_{t_0} &= 180^\circ
 \end{aligned}$$

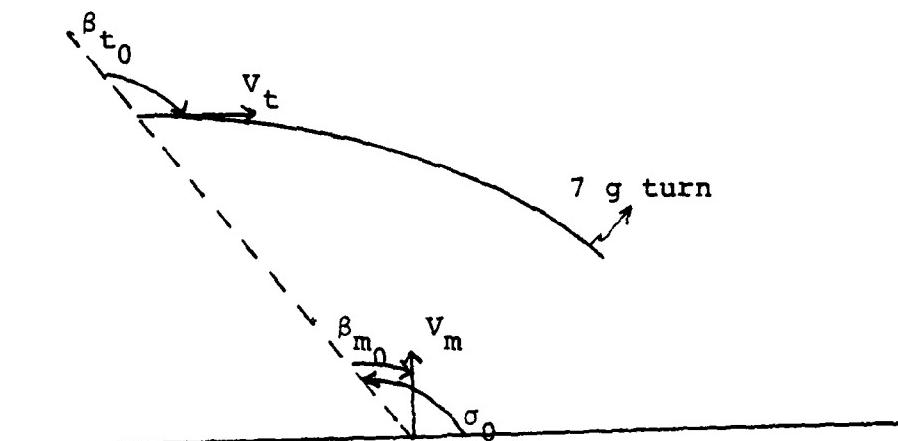
THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

```
INPUT TIME INCREMENT  
INPUT NAVIGATION CONSTANT  
INPUT MISSILE VELOCITY  
INPUT TARGET VELOCITY  
INPUT TARGET ACCELERATION  
INPUT INITIAL MISSILE ACCELERATION  
INPUT INITIAL RANGE  
INPUT BETA TARGET  
INPUT BETA MISSILE  
INPUT THETA TARGET  
INPUT THETA MISSILE  
INPUT SIGMA  
MISSILE POSIT  
TARGET POSIT
```

X1 Y1 X2 Y2
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO

MAXIMUM MISSILE ACC IS 126.1911994 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 9 SEC
INPUT MINIMUM VALUE OF X

Figure (3-23). Head-on scenario.



$$r_0 = 10,000 \text{ m}$$

$$\theta_{t_0} = 0$$

$$v_t = 492 \text{ m/sec}$$

$$\theta_{m_0} = 73.8^\circ$$

$$v_m = 821 \text{ m/sec}$$

$$\sigma_0 = 120^\circ$$

$$\beta_{t_0} = -120^\circ$$

$$k = 4$$

$$\beta_{m_0} = -46.2^\circ$$

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

INPUT TIME INCREMENT
INPUT NAVIGATION CONSTANT
INPUT MISSILE VELOCITY
INPUT TARGET VELOCITY
INPUT TARGET ACCELERATION
INPUT INITIAL MISSILE ACCELERATION
INPUT INITIAL RANGE
INPUT BETA TARGET
INPUT BETA MISSILE
INPUT THETA TARGET
INPUT THETA MISSILE
INPUT SIGMA

MISSILE POSIT

TARGET POSIT

X1 Y1

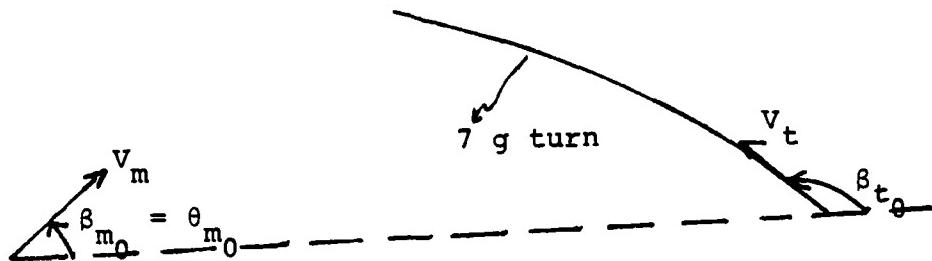
X2

Y2

DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

MAXIMUM MISSILE ACC IS 263.1995227 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 8.8 SEC
INPUT MINIMUM VALUE OF X:

Figure (3-24). Crossing scenario.



$$\begin{array}{ll}
 v_t = 492 \text{ m/sec} & \theta_{t_0} = 135^\circ \\
 v_m = 821 \text{ m/sec} & \theta_{m_0} = 53.9^\circ \\
 \beta_{t_0} = 135^\circ & \sigma_0 = 0 \\
 \beta_{m_0} = 53.9^\circ & k = 4 \\
 r_0 = 10,000 \text{ m} &
 \end{array}$$

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

INPUT TIME INCREMENT
INPUT NAVIGATION CONSTANT
INPUT MISSILE VELOCITY
INPUT TARGET VELOCITY
INPUT TARGET ACCELERATION
INPUT INITIAL MISSILE ACCELERATION
INPUT INITIAL RANGE
INPUT BETA TARGET
INPUT BETA MISSILE
INPUT THETA TARGET
INPUT THETA MISSILE
INPUT SIGMA
MISSILE POSIT TARGET POSIT

X1 Y1 X2 Y2
DO YOU WANT A PRINT OF THE OUTPUT, 0=YES, 1=NO

MAXIMUM MISSILE ACC IS-156.9795962 METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 8.8 SEC
INPUT MINIMUM VALUE OF X

Figure (3-25). Oblique scenario.

TABLE 3-IX

INPUT TIME INCREMENT			
INPUT NAVIGATION CONSTANT			
INPUT MISSILE VELOCITY $\rightarrow M_M = 2.0$			
INPUT TARGET VELOCITY			
INPUT TARGET ACCELERATION			
INPUT INITIAL MISSILE ACCELERATION			
INPUT INITIAL RANGE			
INPUT BETA TARGET			
INPUT BETA MISSILE			
INPUT THETA TARGET			
INPUT THETA MISSILE			
INPUT SIGMA			
MISSILE POSIT			TARGET POSIT
X1	Y1	X2	Y2

DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO

MAXIMUM MISSILE ACC IS 275.5734145	METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 10.3	SEC
INPUT MINIMUM VALUE OF X	

THIS PROGRAM FINDS THE MAXIMUM
MISSILE ACCELERATION FOR A
PROPORTIONAL NAVIGATION SYSTEM

INPUT TIME INCREMENT			
INPUT NAVIGATION CONSTANT			
INPUT MISSILE VELOCITY $\rightarrow M_M = 3.0$			
INPUT TARGET VELOCITY			
INPUT TARGET ACCELERATION			
INPUT INITIAL MISSILE ACCELERATION			
INPUT INITIAL RANGE			
INPUT BETA TARGET			
INPUT BETA MISSILE			
INPUT THETA TARGET			
INPUT THETA MISSILE			
INPUT SIGMA			
MISSILE POSIT			TARGET POSIT
X1	Y1	X2	Y2

DO YOU WANT A PRINT OF THE OUTPUT, 0=YES,1=NO

MAXIMUM MISSILE ACC IS 248.233583	METERS/SEC/SEC
MISSILE TIME OF FLIGHT IS 7.8	SEC
INPUT MINIMUM VALUE OF X	

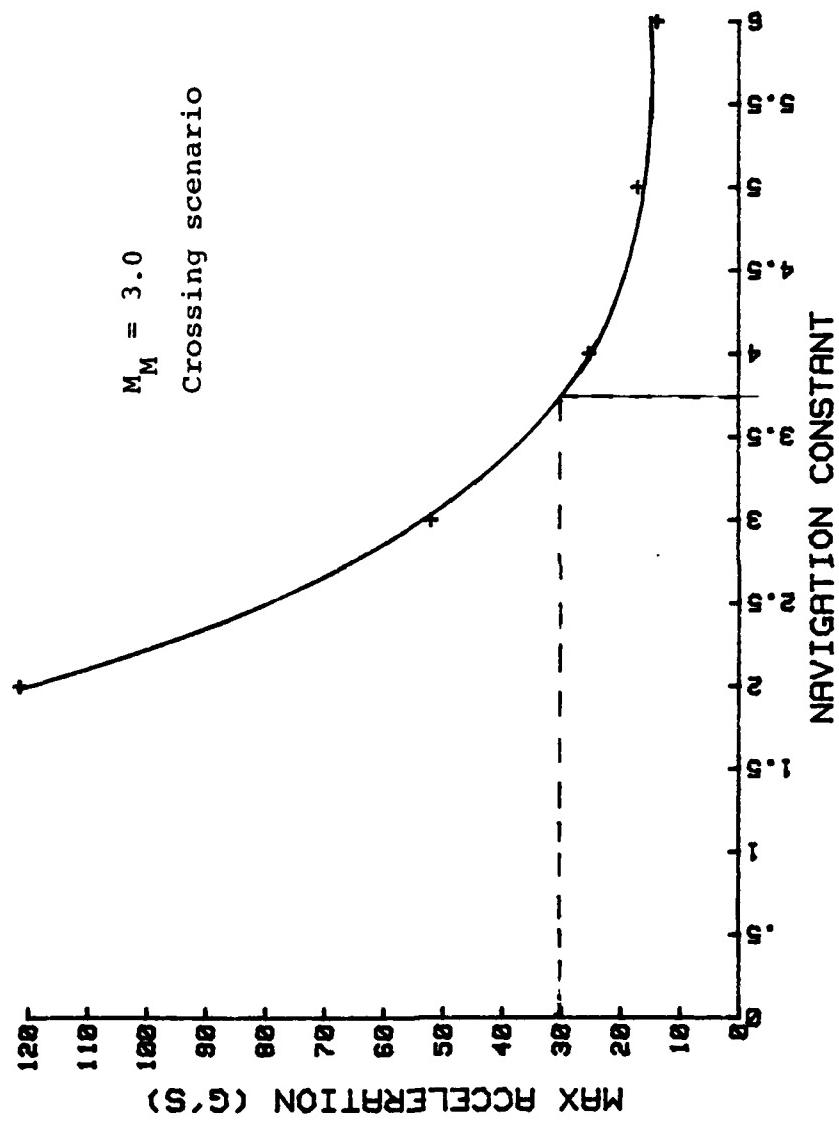


Figure (3-26). Selection of Navigation Constant.

acceleration decreased. The results for $M_M = 2.0$ and $M_M = 3.0$ are shown in Table 3-IX. This result indicates the desirability of retaining the missile Mach number originally selected.

For the crossing case and a missile speed of $M_M = 3.0$, the proportionality constant was then varied from $k = 2$ to $k = 6$. The results are plotted in Figure (3-26). If the maximum acceleration is limited to 31 g's ($3a_t + 10$), the required proportionality constant is $k = 3.75$. This is well within the desirable range of 2 - 6 indicated earlier.

From this analysis the required performance objectives are:

$$M_M = 3.0$$

$$k = 3.75$$

$$(a_m)_{\max} = 31 \text{ g's}$$

IV. SIZING THE DIAMETER

The missile diameter is determined by one of three driving factors. For relatively short range missiles the diameter will be fixed by either the warhead or the seeker requirements. As might be expected, for longer range missiles the diameter will more likely be fixed by either the warhead or the seeker requirements. As might be expected, for longer range missiles the diameter will more likely be fixed by the propulsion requirements in order to prevent excessive propulsion system lengths. An initial estimate of the missile diameter must be made at this point in order to proceed with the design. The initial seeker requirement can be determined from a knowledge of the lock-on range requirement found in Chapter 2. The warhead necessary to inflict a "kill" can also be estimated from information about the target and characteristic explosives. The propulsion requirement cannot be determined because of the lack of any aerodynamic drag or weight information at this point. For this reason the missile diameter will be now sized for seeker or warhead requirements. The missile propulsion requirements will be determined later in the design process, and it may be necessary at that time to resize the missile to meet these propulsion requirements.

Selection of the type of seeker depends upon the operational arena of the missile. The seeker of a shoulder fired, battlefield missile would not be the optimum seeker of a shipboard

missile where antenna and component sizes are not limiting factors. All types of guidance use some portion of the electromagnetic spectrum. The three primary areas of use are the electro-optical, infrared, and radio frequencies. The millimeter wave section of the spectrum is also of current interest in the design of missiles due to small component size and will also be discussed. The following table lists some of the major advantages and disadvantages of the three.

	<u>Advantages</u>	<u>Disadvantages</u>
Optical	Target resolution (detail) Real time information Three dimension effect	Bad weather degrades Night use degrades
Infrared	Improved resolution over RF	Attenuation due to aerosols and atmosphere
RF	Longest range Least absorption and attenuation	Larger components

A. THE RADAR RANGE EQUATION

An omnidirectional antenna is one that radiates power in all directions equally. If the power radiated by an antenna is P_t , the power density at a distance R_t from the source is given by,

$$\text{Power density} = P_t / (4\pi R_t^2)$$

$$4\pi R_t^2 = \text{area of a sphere of radius } R_t$$

Since antennas are normally directive instead of omnidirectional, most of the power is radiated in a particular direction. The

gain, G_t , is a measure of the increased power from a directive antenna as compared to an omnidirectional antenna. Therefore the power density from a directive antenna can be expressed as,

$$\text{Power density} = \frac{P_t G_t}{4\pi R_t^2}$$

This is the power density which arrives at the target. The target intercepts a portion of this energy and reradiates it in the opposite direction. The radar cross section, σ , is a measure of the effective area of the target. The power radiated by the target is P_{echo} ,

$$P_{\text{echo}} = \frac{P_t G_t \sigma}{4\pi R_t^2}$$

This energy propagates as if it were radiated by an omnidirectional antenna. Therefore if the receiving antenna is a distance, R_r , away the power density at the receiver is

$$(\text{Power density})_r = \frac{P_t G_t \sigma}{(4\pi R_t^2)(4\pi R_r^2)}$$

If the energy is intercepted by the receiving antenna, which has an effective area as seen by the returned energy of A_r ; then the power received by the radar, P_r , is

$$P_r = \frac{P_t G_t \sigma A_r}{(4\pi R_t^2)(4\pi R_r^2)} \quad (1)$$

This is the simplest form of the radar equation and can be used to determine the size of antenna required.

B. ACTIVE RADAR HOMING

Active homing is the method of missile guidance in which the radar transmitter and receiver are located on-board the missile. In this case the same antenna is used for both transmitting and receiving. The radar equation then becomes,

$$P_r = \frac{P_t G_t \sigma A_t}{(4\pi R^2)^2}$$

where, $R_t = R_r = R$ and $A_r = A_t$.

The minimum power for which the target can be detected, P_{min} , is a function of many variables. A full development of this term can be found in reference (8).

$$P_{min} = k T_0 B_n F_n \left(\frac{S_0}{N_0}\right)_{min} .$$

Boltzmann's constant is $k = 1.38 \times 10^{-23}$ joule/ $^{\circ}$ K. The value of kT_0 at room temperature is 4×10^{-21} watt/cps of bandwidth. The bandwidth, B_n , noise figure, F_n , and minimum signal to noise ratio, $(S_0/N_0)_{min}$ are all functions of the receiver. Typical

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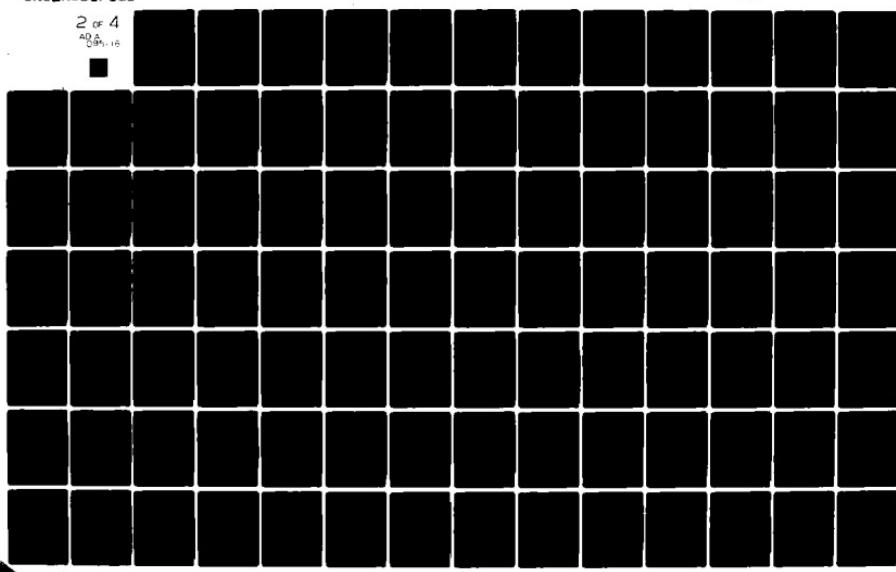
NAVAL POSTGRADUATE SCHOOL MONTEREY CA
TACTICAL MISSILE CONCEPTUAL DESIGN, (U)
SEP 80 D R REDMON

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values are listed below.

$$B_n = 1 \text{ MHz}$$

$$F_n = 7.5 \text{ db} = 2.37 \text{ (for crystal mixer)}$$

$$(S_0/N_0)_{\min} = 14.7 \text{ db} = 5.43 \text{ (for probability of detection, } P_D = 0.9 \text{ and probability of false alarm, } P_{fa} = 1/15 \text{ minutes)}$$

The above values give $P_{\min} = 5.15 \times 10^{-14}$ watts. This is the value which will be used throughout this section.

From antenna theory the gain is related to the effective antenna area by,

$$G = \frac{4\pi A}{\lambda^2}$$

The maximum radar range can then be shown to be

$$R_{\max} = \left[\frac{P_t \sigma A_t^2}{4\pi \lambda^2 P_{\min}} \right]^{1/4} \quad (1)$$

Equation (1) for radar range does not include any system losses. It also does not include the statistical nature of several of the parameters. Because of these assumptions the actual range of a radar may be as small as one-half of what the radar range equation predicts for laboratory conditions. For this reason, twice the required range should be used when using the above equation.

1. Example

The AN/APQ-153 is the airborne attack radar system used on the F-5E aircraft. The following parameters apply to this radar.

$$f_0 = 8-10 \text{ GHz}$$

$$\lambda = c/f_0 = \frac{3 \times 10^8 \text{ m/sec}}{9 \times 10^9 / \text{sec}} = .0333 \text{ m}$$

$$c = \text{speed of electromagnetic propagation}$$
$$= 3 \times 10^8 \text{ m/sec}$$

$$P_t = 80 \text{ kW}$$

Antenna = Parabolic dish 30.5 x 40.6 cm

$$A_t = 0.12383 \text{ m}^2$$

for a target of 1 square meter of radar cross-section,

$$R_{\max} = 36.16 \text{ km} = 19.5 \text{ nmiles}$$

For the case of an active homing radar, the size of the transmitting and receiving antenna is the parameter of interest. The antenna size may very well drive the design diameter of the missile. The antenna diameter can be expressed as follows.

$$A_t = \left[\frac{4\pi \lambda^2 P_{\min} R_{\max}^4}{P_t \sigma} \right]^{1/2}$$

Let $C = \left[\frac{4\pi \lambda^2 P_{\min}}{P_t \sigma} \right]^{1/2}$ (2)

then

$$A_t = \frac{\pi d_t^2}{4} = C R_{\max}^2$$

$$d_t^2 = \frac{4C R_{\max}^2}{\pi} \quad \text{therefore, } d_t = 2 R_{\max} \sqrt{\frac{C}{\pi}}$$

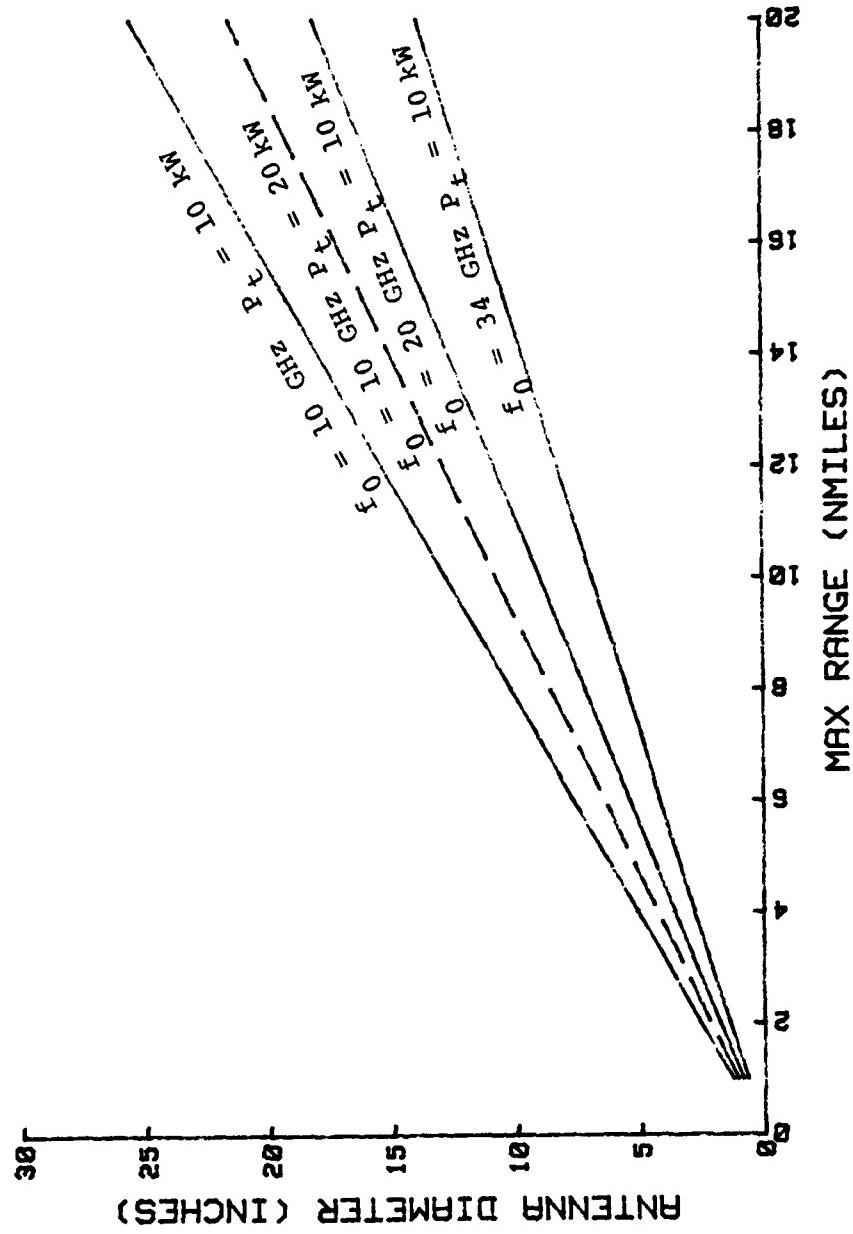


Figure (4-11). Required antenna size.

Equation (2) has been plotted in Figure (4-1) for various values of the transmitter frequency. From this plot and a knowledge of the maximum lock-on range required, the antenna size can be determined.

From equation (2) and Figure (4-1) there are two obvious ways to decrease the antenna size required. (1) Increasing frequency is the best way to reduce antenna and electronic component sizes. The current trend is toward higher frequencies. (Millimeter waves.) One problem is that the equations developed in this section do not include atmospheric attenuation. For frequencies above about 30 GHz the absorption due to atmospheric gases increases. This is shown on Figure (4-2). As indicated on this figure there are "windows" where the attenuation is less. These "windows" occur at frequencies of 34 GHz, 94 GHz, 140 GHz, and 220 GHz. These are the frequencies where most of the current research and development is going on. As the frequency increases, the wavelength approaches the size of rain droplets. For this reason, radar performance is greatly reduced in inclement weather. (2) Increasing transmitter power will also decrease the size of antenna necessary. The limiting factor in this area is the lack of high power sources. In the millimeter range the available power from current traveling wave tubes is 50-100 watts. Increasing power is obviously confined to size and weight limitations of missile components.

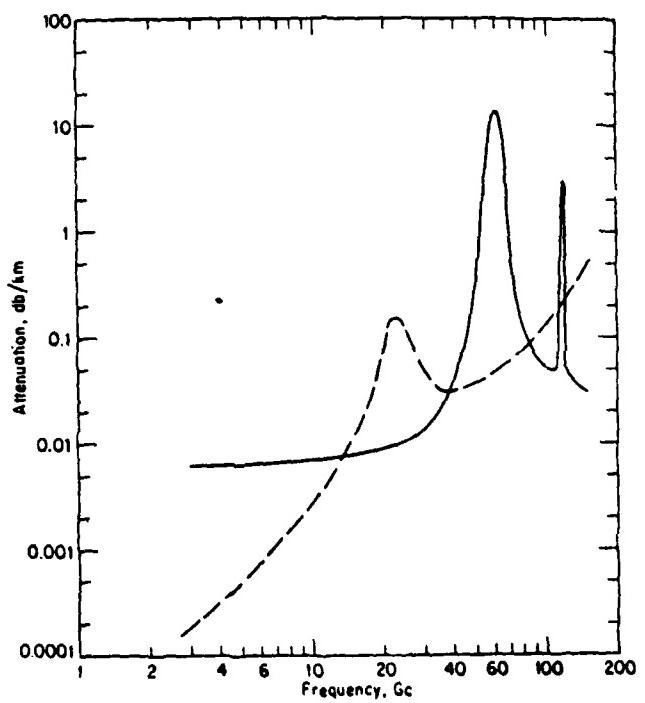


Figure (4-2). Atmospheric Absorption [8].

C. SEMI-ACTIVE HOMING

The advantage of semi-active homing is obvious when the radar range equation is investigated. From equation (1),

$$R_t^2 R_r^2 = \frac{P_t G_t \sigma A_r}{(4\pi)^2 P_{min}}$$

In the above equation the missile range from the target is R_r . The transmitting and receiving antennas are at different ranges and have different characteristics in this case. As before,

$$G_t = \frac{4\pi A_t}{\lambda^2}$$

therefore,

$$R_t^2 R_r^2 = \frac{P_t A_t A_r \sigma}{4\pi \lambda^2 P_{min}}$$

The main advantage is in the transmitter characteristics. Since the transmitter is not located in the missile, it is not normally limited in size and weight requirements. In the above equation if a transmitter power and standoff range, R_t , is chosen the receiving antenna can be sized for a maximum homing range, R_{max} , of the missile.

1. Example

$$P_t = 100 \text{ km}$$

$$R_t = 100 \text{ nmiles}$$

$$f_0 = 10 \text{ GHz}$$

$$A_t = 4 \text{ m}^2$$

$$P_{min} = 5.15 \times 10^{-14} \text{ W}$$

<u>R_{max} (nmiles)</u>	<u>d_r (in)</u>
10	5.8
20	11.6
50	29.1

It can be seen from comparing these numbers to those of Figure (4-1) that the required antenna size is less than one half of that required for an active homing radar of the same frequency.

D. DESIGN EXAMPLE (ANTENNA SIZING)

An active radar was assumed in Chapter 2 to decrease the required missile range. The lock-on range was 10,000 m or 5.4 nmiles. As stated earlier, twice this number should be used for determining antenna size. From Figure (1) for a range of 10.8 nmiles, and a transmitter power of 10 kW at $f_0 = 20$ GHz, the required antenna size is $d = 10$ inches.

E. INFRARED SEEKERS

In the design of missile seekers two parameters of primary importance are range and size. The idealized range for an infrared tracker relates these two factors. The idealized range is the range at which the signal-to-noise ratio is unity and is given by,

$$R_0 = \left[\frac{D^* T_a T_{IR} D_a^2 J}{4 \sqrt{\Delta f A_d}} \right]^{1/2} \quad (1)$$

		<u>Typical value</u>
Specific detectivity	D*	10^{10}
Transmission through atmosphere	T_a	0-1.0
Transmission through IR optics	T_{IR}	0-1.0
Aperture diameter	D_a	
Radiant intensity	J	10^3
Receiver bandwidth	Δf	10^3
Sensitive area of detector	A_d	$10^{-6} - 10^{-1} \text{ cm}^2$

The derivation of equation (1) and its use are the subjects of this section. Some references (9) may give the above equation in terms of the Noise Equivalent Intensity, NEI.

$$R_0 = \left[\frac{J}{NEI} \right]^{1/2}$$

where

$$NEI = \frac{\sqrt{\Delta f} A_d}{D^* A_a T_a TIR}$$

1. Planck's Law

The radiant emittance of a body is a measure of the radiant power per unit area emitted from the surface.

$$W = \frac{P}{A} \text{ watt/cm}^2 \quad (2)$$

The spectral radiant emittance is the radiant emittance per unit wavelength interval,

$$W_\lambda = \frac{\partial W}{\partial \lambda} \text{ watt/cm}^2 \mu$$

$\mu = \text{micron} = 10^{-6} \text{ meters.}$

Planck's law gives the blackbody spectral radiant emittance as a function of wavelength and temperature,

$$(W_\lambda)_{BB} = \frac{2 \pi h c^2}{\lambda^5} \frac{1}{\exp(hc/\lambda kT) - 1} \quad (3)$$

$h = \text{Planck's constant} = 6.6238 \times 10^{-34} \text{ Joule-sec}$

$c = \text{speed of light} = 3 \times 10^8 \text{ m/sec}$

$\lambda = \text{wavelength}$

$k = \text{Boltzmann's constant} = 1.38 \times 10^{-23} \text{ Joule/}^\circ\text{K}$

$T = \text{Absolute temperature, } ^\circ\text{K}$

Figure (4-3) shows equation (3) for various absolute temperatures. As can be seen the wavelength at which maximum radiant emittance occurs varies with temperature. This maximum occurs at a wavelength given by Wien's displacement law, λ_{\max}

$$\lambda_{\max} T = 2897.8 \mu \text{ } ^\circ\text{K}$$

2. Emissivity

Actual bodies do not emit radiation according to Planck's law. A more typical plot of radiant emittance is shown in Figure (4-4). Spectral emissivity, ϵ_λ , is defined as the ratio of the actual spectral radiant emittance to the blackbody spectral radiant emittance,

$$\epsilon_\lambda = \frac{W_\lambda}{(W_\lambda)_{BB}}$$

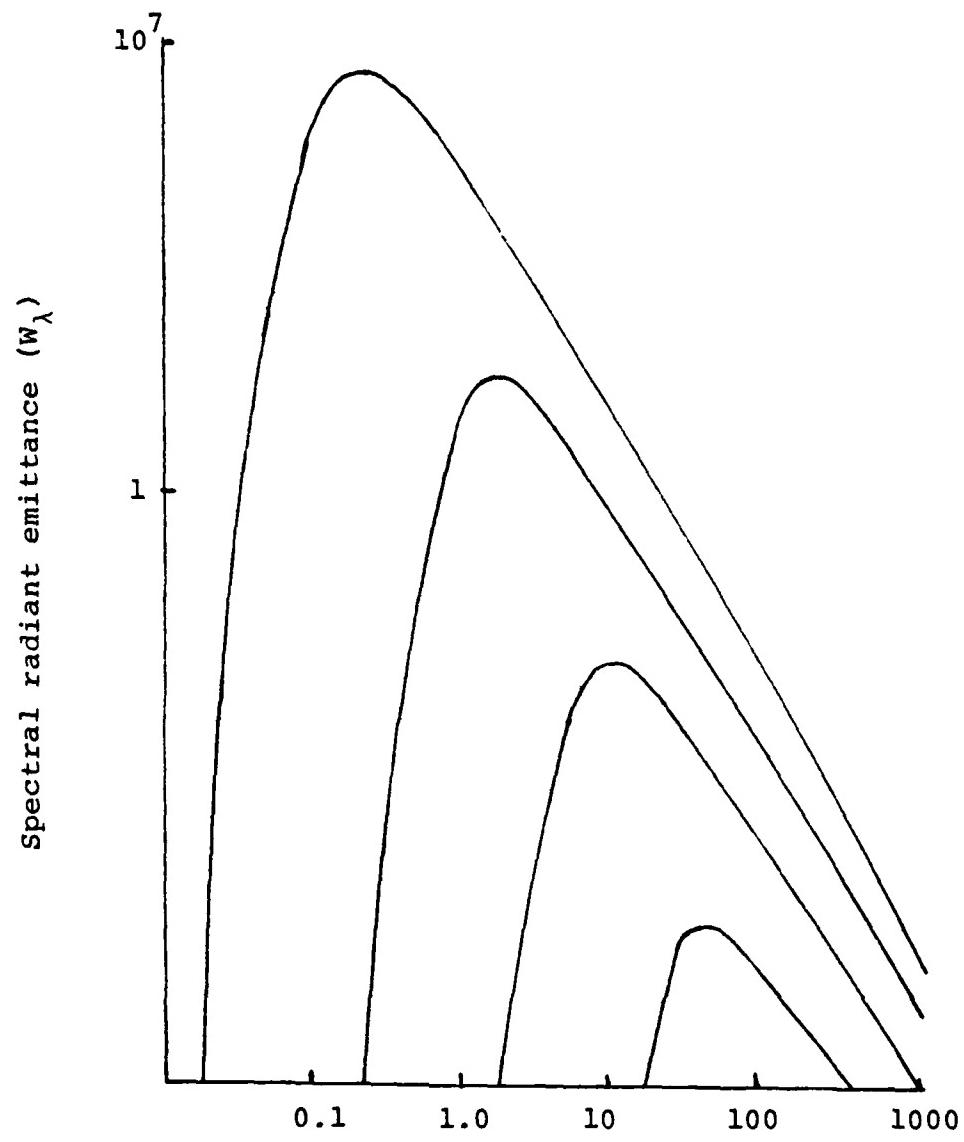


Figure (4-3). Spectral radiant emittance.

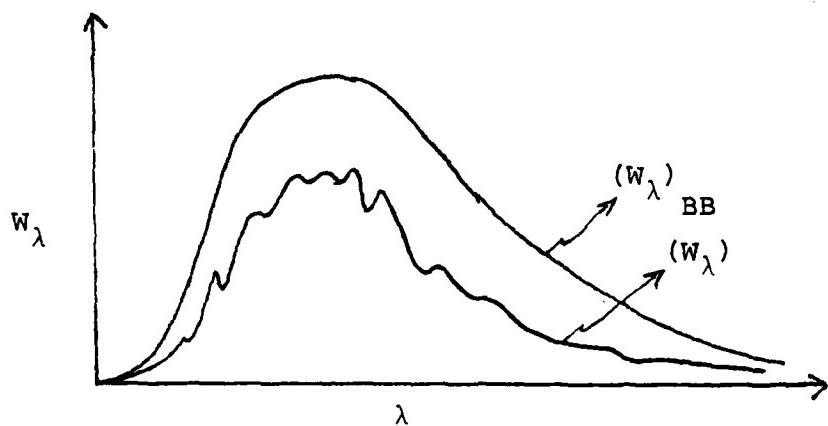


Figure (4-4). Actual spectral radiant emittance.

As shown in the above figure, ϵ_λ , may vary with wavelength. A grey body is defined as one which has constant spectral emissivity,

$$\epsilon = \epsilon_\lambda = \text{constant}$$

IR systems normally use filters to limit the accepted radiation to a specific wavelength band. The radiant emittance of a body between wavelengths λ_1 and λ_2 becomes,

$$W = \int_{\lambda_1}^{\lambda_2} \epsilon(W_\lambda)_{BB} d\lambda$$

In Figure (4-5) the surface at the origin emits a total energy WA into a hemisphere normal to A .

The radiance is defined as the radiant power per unit solid angle per unit projected area,

$$\text{Radiance } N = \frac{\partial^2 P}{\cos\theta \partial A \partial \Omega} = \frac{1}{\cos\theta} \frac{\partial}{\partial \Omega} \frac{\partial P}{\partial A}$$

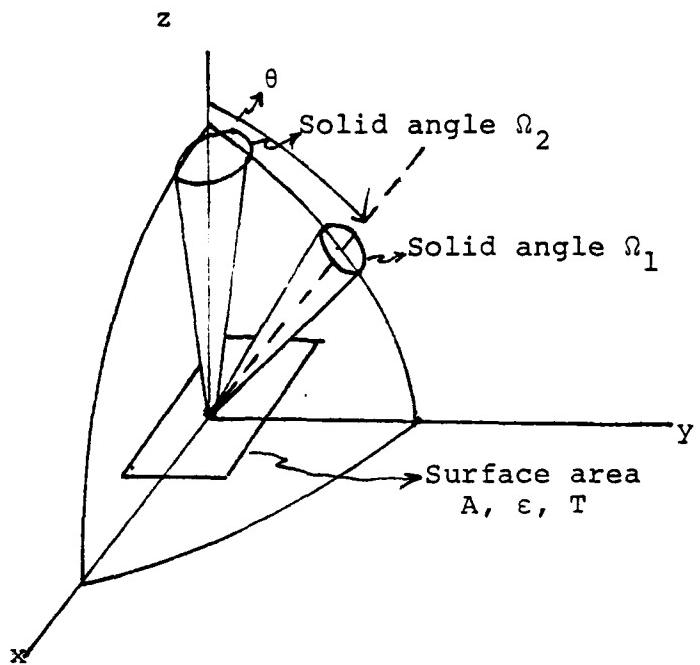


Figure (4-5). Radiant emittance of a body.

From equation (2)

$$N = \frac{1}{\cos\theta} \frac{\partial W}{\partial\Omega} \text{ watts/steradian cm}^2$$

The radiant intensity is defined as the radiant power per unit solid angle from a point source.

$$\text{Radiant intensity } J = \frac{\partial P}{\partial\Omega} \text{ watt/steradian}$$

The following is a summary of the definitions of radiant energy quantities,

W = radiant emittance, watts/cm²

J = radiant intensity, watts/steradian

N = radiance, watts/steradian-cm²

From the definitions the relationship between W , J and N are,

$$W = \pi N = \pi J/A$$

From the above definitions and Figure (4-5) the energy into solid angle, Ω_1 is,

$$NA\Omega_1 \cos \theta = \frac{W}{\pi} \Omega_1 A \cos \theta$$

The energy into solid angle Ω_2 is given by,

$$NA\Omega_2 = \frac{W}{\pi} \Omega_2 A$$

3. Energy into a Hemisphere

From the definition of radiance and the radiant emittance, the radiance in terms of the radiant emittance can be found,

$$N = \frac{\partial^2 P}{\cos \theta \partial A \partial \Omega}$$

$$W = \frac{\partial P}{\partial A}$$

$$N = \frac{1}{\cos \theta} \frac{\partial W}{\partial \Omega}$$

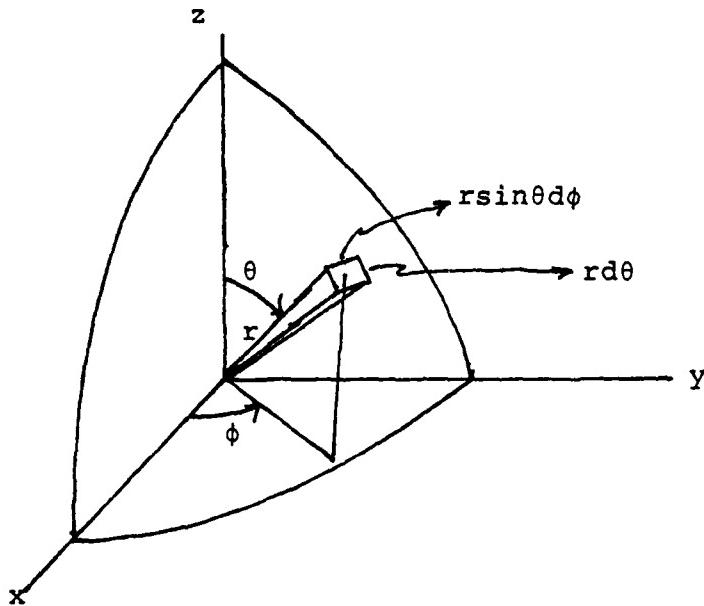


Figure (4-6). Energy emitted into a hemisphere.

From Figure (4-6) the incremental solid angle is

$$d\Omega = \frac{r \sin \theta d\phi r d\theta}{r^2} = \sin \theta d\theta d\phi$$

If the above surface, A_t , is considered a Lambertian surface, the radiance, N , is independent of the direction of radiation.

$$dW = \cos \theta N d\Omega$$

$$dW = \cos \theta N \sin \theta d\theta d\phi$$

The total radiant emittance into a hemisphere above the surface is then,

$$W = \int_0^{2\pi} \int_{-\pi/2}^{\pi/2} N \cos \theta \sin \theta d\theta d\phi$$

$$W = 2\pi N \left[\frac{1}{2} \sin^2 \theta \right]_0^{\pi/2}$$

$$W = \pi N$$

4. Targets

Infrared targets include a wide variety of radiation sources. The radiance of most bodies can be divided into that due to self-emission and that due to reflection of incident radiation,

$$N = N_e + N_r$$

The relative magnitude of these contributions depends on a number of factors and varies from target to target and operating environment.

a. Self-emission.

Self-emission, also referred to as thermal emission, depends primarily on the temperature of the body and the emissivity. The most often used approach is to consider the body as a grey body which emits radiation according to the Stefan-Boltzmann law.

$$N_e = \frac{\epsilon \sigma T^4}{\pi} \text{ watts/steradian-cm}^2$$

where $\sigma = 5.67 \times 10^{-12} \text{ watts/cm}^2 (\text{°K})^4$

This term is the total radiance (over all wavelengths) and is not the same as used previously.

b. Reflection.

Radiance due to reflectance depends on the illuminating source. This source may be the sun, active, or semi-active sources. It is obvious that at night for a passive infrared system, the radiance due to reflection is not a contributing factor. For this reason only the radiance due to self-emission is considered in this section.

5. Target Temperature

Since the self-emittance of a target depends on the temperature of the target, a method for determining this temperature is needed. The temperature of an aerial target varies depending on the aspect of the target. The propulsion system has hot surfaces such as the nozzle and exhaust plumes. There may also be hot surfaces due to aerodynamic heating and/or solar radiation.

Air breathing engines normally have exhaust plumes ranging from 600 to 1000°K. Rockets typically have much hotter plumes. The flame temperatures for liquid propellants range from 2500 to 7500°K. Solid propellants flame temperatures range from 1700 to 3500°K. The plume temperature can be estimated from the relation.

$$\frac{T_{\text{flame}}}{T_{\text{plume}}} = \frac{T_0}{T_e} = 1 + \frac{\gamma-1}{2} M_e^2$$

where T_0 = stagnation temperature

T_e = static temperature

M_e = Mach number at the nozzle exit

a. Example I

For a flame temperature of 2700°K and an exit Mach number of 3.0 the plume temperature can be found,

$$T_{\text{plume}} = \frac{T_{\text{flame}}}{1 + \frac{\gamma-1}{2} M_e^2} = 1270.6^\circ\text{K}$$

For many missile encounters the exhaust plume may be shielded from the infrared sensor. For a head-on encounter the temperature of interest is the skin temperature of the target. This temperature is due to aerodynamic heating and is a function of the target speed and the target material. One approach to finding this temperature is through the use of the recovery

factor, which requires some knowledge of the material of the target. The recovery factor, r , of a material is defined as follows:

$$r = \frac{T_{\text{surface}} - T_{\text{ambient}}}{T_{\text{stagnation}} - T_{\text{ambient}}}$$

The skin temperature of the target then becomes,

$$T_{\text{surface}} = T_{\text{ambient}} + r (T_{\text{stag}} - T_{\text{amb}})$$

The stagnation temperature is found from the relation,

$$T_{\text{stag}} = T_{\text{amb}} \left(1 + \frac{\gamma-1}{2} M^2\right)$$

The Mach number, M , is that of the target and the specific heat ratio, γ , is for air.

b. Example II

A target flying at $M = 2.5$ where the ambient temperature is 300°K has a recovery factor of 0.75.

$$T_{\text{stag}} = 575^{\circ}\text{K}$$

$$T_{\text{surface}} = 581^{\circ}\text{K}$$

This is the temperature used, along with the emissivity of the target, to find the radiant emittance of the target from equation (4).

6. Simple IR System

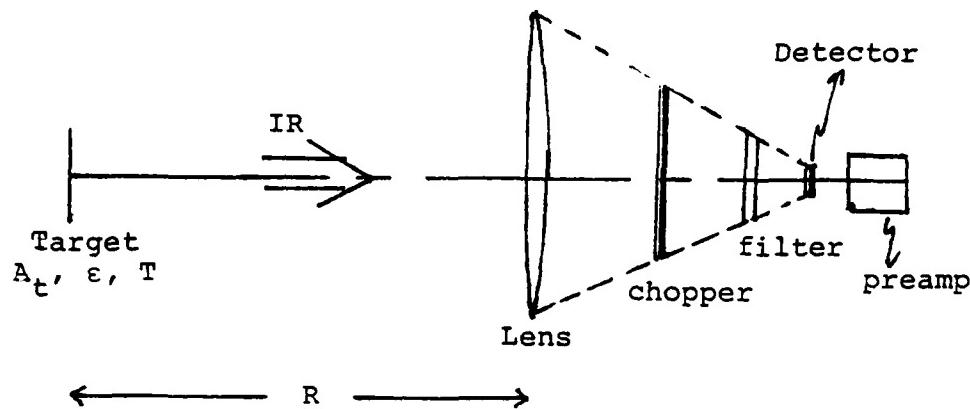


Figure (4-7). Simple IR system.

Figure (4-7), above, shows a simple IR system and a target at a range R. If the system is sensitive to radiation in the 3 to 5 micron region the radiant emittance becomes,

$$W = \int_{\lambda_1=3}^{\lambda_2=5} \epsilon (W_\lambda)_{BB} d\lambda$$

$$W = \epsilon \int_3^5 \left[\frac{2\pi hc^2}{\lambda^5} \frac{1}{\exp(hc/kT) - 1} \right] d\lambda$$

The above integral is best evaluated on the computer. If the ambient temperature and target speed is known, the skin temperature of the target can be determined.

The radiance from the target then becomes,

$$N = \frac{W}{\pi}$$

If Figure (4-7), the solid angle of the aperture as seen from the target is, $\Omega = \frac{\pi a^2}{R^2}$

The power seen at the detector surface is then,

$$\text{IR Power} = N \Omega A_t$$

The above formula assumes no attenuation by the atmosphere or the IR system optics. This attenuation is significant in actual IR systems. These factors are normally accounted for through the use of atmospheric and IR optics transmission coefficients.

T_a = transmission of the atmosphere

T_{IR} = transmission of the IR optics

The total power at the detector then becomes,

$$P = \frac{T_a T_{IR} A_t A_a W}{\pi R^2} \quad (5)$$

7. Detectors

Detectors are devices which are radiation transducers. It's purpose is to change the incoming radiant power to an electrical signal, which can then be amplified. Detectors can be divided into two main categories. (1) Thermal detectors - The responsive element of a thermal detector is sensitive to

temperature changes brought about by the incident radiation.

(2) Photodetectors - Responsive elements of photodetectors are sensitive to the number of incident photons.

Detectors also are made up of windows, apertures and Dewar flasks. The window restricts the bandwidth to which the detector is sensitive. The aperture may limit the field of view in order to limit photon noise. The Dewar flask cools the detector which improves the detectivity.

Detectivity of a detector is defined as,

$$D = \frac{\text{signal/noise}}{\text{input power}} = \frac{S/N}{P} \quad (6)$$

The specific detectivity is

$$D^* = D [\Delta f A_d]^{1/2} \quad (7)$$

Δf = Bandwidth

A_d = Sensitive area of detector

For a tracking system the bandwidth is that of the preamplifier in Figure (4-7). The input to the preamplifier is proportional to the incoming IR energy, which has been modulated to give target resolution from the background and provide line-of-sight information.

A simple chopper is shown in Figure (4-8). It consists of an opaque material which has a wedge cut-out of angle α . The rotation causes the input from a point source to be modulated,

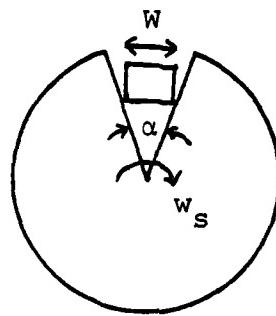


Figure (4-8). Simple Chopper.

while that of the background is not. The input signal to the preamplifier would look like Figure (4-9). The frequency content of the signal in Figure (4-9) can be found from a Fourier Analysis. If the pulses are assumed to be sinusoidal of period T , the optimum bandwidth is,

$$\Delta f = \frac{3}{T}$$

$$T = \frac{W}{\pi \omega w_s}$$

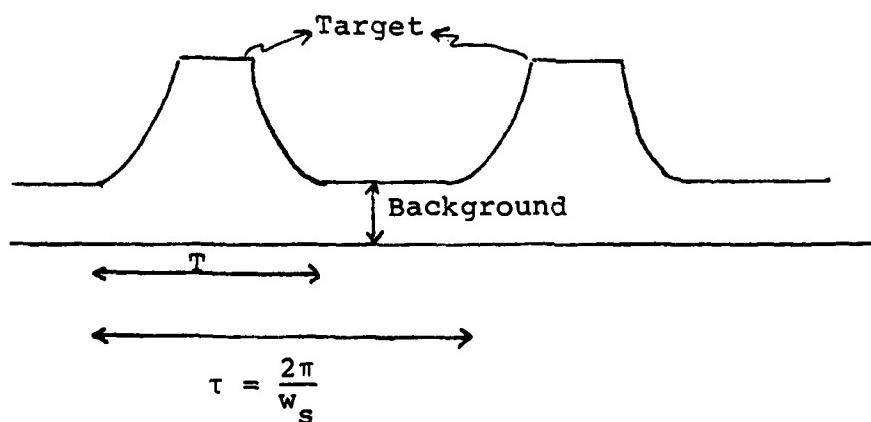


Figure (4-9). Preamp input.

The specific detectivity is characteristic of the detector used. Reference [9] is an excellent source of information on operational detectors.

8. Idealized Range

Equation (5) is,

$$P = \frac{T_a T_{IR} A_t A_a W}{\pi R^2}$$

From equations (6) and (7) this becomes,

$$\frac{D^*}{[\Delta f A_d]}^{1/2} = \frac{S/N \pi R^2}{T_a T_{IR} A_t A_a W}$$

The idealized range, where the signal to noise ratio is unity, is

$$R_0^2 = \frac{D^* T_a T_{IR} A_t A_a W}{[\Delta f A_d]^{1/2} \pi}$$

To simplify this equation the radiant intensity is given by,

$$J = \frac{A_t W}{\pi}$$

Since the parameter of interest in missile design is the aperture diameters A_a , it is replaced with $\frac{\pi D_a^2}{4}$ so that,

$$R_0 = \left[\frac{D^* T_a T_{IR} \pi D_a^2 J}{4 \sqrt{\Delta f A_d}} \right]^{1/2}$$

$$D_a = \left[\frac{R_0^2 4 \sqrt{\Delta f A_d}}{D^* T_a T_{IR} \pi J} \right]^{1/2}$$

a. Example (Idealized Range)

From example II a target flying at $M = 2.5$ has a skin temperature of 581°K . If this target has a presented area of 1 m^2 , the detector size needed to detect the target at a range of 10,000 m can be determined. From Wien's displacement law the maximum radiation occurs at,

$$\lambda_{\max} = 4.99 \mu$$

If the system is designed to accept radiation from 3 to 5 microns, and the emissivity of the target is 0.7.

$$W = \epsilon \int_{3}^{5} \frac{2\pi hc^2}{\lambda^5} \frac{1}{\exp(hc/\lambda kT) - 1} d\lambda$$

If this equation is integrated on the computer, the radiant emittance becomes,

$$W = 1410 \text{ watts/m}$$

The radiant intensity becomes,

$$J = \frac{A_t W}{\pi} = 448.82 \text{ watts}$$

Typical values of the parameters in the idealized range equation are (9)

$$T_a = 0.75$$

$$T_{IR} = 0.95$$

$$\Delta f = 1000 \text{ cps}$$

$$A_d = 1 \text{ cm}^2$$

$$D^* = 1 \times 10^{10} (\text{cps})^{1/2} \text{ cm/watt}$$

Substituting these values into the idealized range equation gives,

$$D_a = 0.0355 \text{ m} = 1.4 \text{ in.}$$

As can be seen the size of the IR seeker is relatively small compared to other seekers. The above analysis is for the "idealized" range. Attenuation of IR radiation can be quite high thereby increasing the seeker size required.

F. WARHEAD SIZING

The conditional kill probability of a missile is the probability that the target is destroyed given that the warhead is delivered to a point in space and the fuze detonates the warhead at a miss distance r .

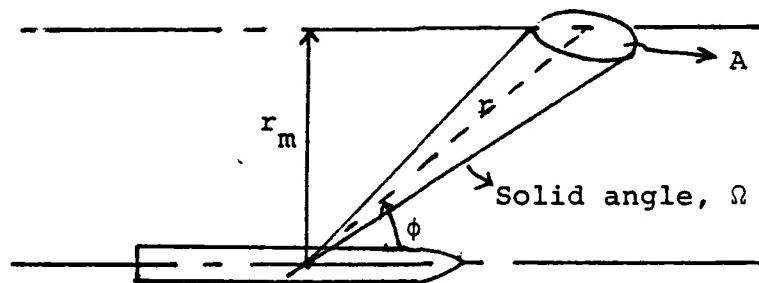


Figure (4-10). Encounter geometry.

The fragment distribution, $D(\phi)$, is the number of fragments per unit solid angle, Ω . The total number of fragments within the cone is,

$$N = D(\phi)\Omega$$

The fragment density, ρ , is the number of fragments per unit of area normal to the path.

$$\rho = \frac{N}{A} = \frac{D(\phi)\Omega}{A}$$

however,

$$\Omega = \frac{A}{r^2}$$

and

$$\rho = \frac{D(\phi)}{r^2} = \frac{D(\phi) \sin^2 \phi}{r_m^2} \quad (8)$$

1. Target Vulnerability

The vulnerability of aircraft or missile components is normally determined experimentally. Fragments of a specified size are fired at the component, and the damage is assessed to determine if the fragment would cause a kill. If a large number of fragments are fired, the ratio of killing fragments to hits can be determined. This ratio is defined as the probability that given a hit a kill will result, $P_{K/H} = \frac{N_K}{N_H}$. Assuming the distribution of hits is uniform over the target,

$$P_{K/H} \equiv \frac{A_v}{A_p} \quad (9)$$

A_p = presented area of target

A_v = vulnerable area of target

As would be expected, $P_{K/H}$, depends greatly upon the encounter geometry. It will be dependent upon the aspect of the aircraft, and also depends upon the type of kill specified. If the target is assumed to be a spherical target, the probability of kill given a hit can be assumed constant.

2. Conditional Kill Probability

The fragment density is given by equation (8). From this density expression, the average number of hits, a , on the vulnerable area of the target is,

$$a = \frac{D(\phi) \sin^2 \phi}{2} \frac{A_v}{r_m}$$

It is customary to assume that the distribution of hits on the presented area follows a Poisson distribution. The conditional kill probability then becomes,

$$P_D = 1 - e^{-a} \quad (10)$$

$$P_D = 1 - \exp \left[\frac{-D(\phi) \sin^2 \phi A_v}{2 r_m^2} \right]$$

The above expression depends upon the fragment distribution, $D(\phi)$. If the warhead casing is scored such that it produces N fragments of uniform size and mass, m , the problem is simplified by formulating an alternate expression for a .

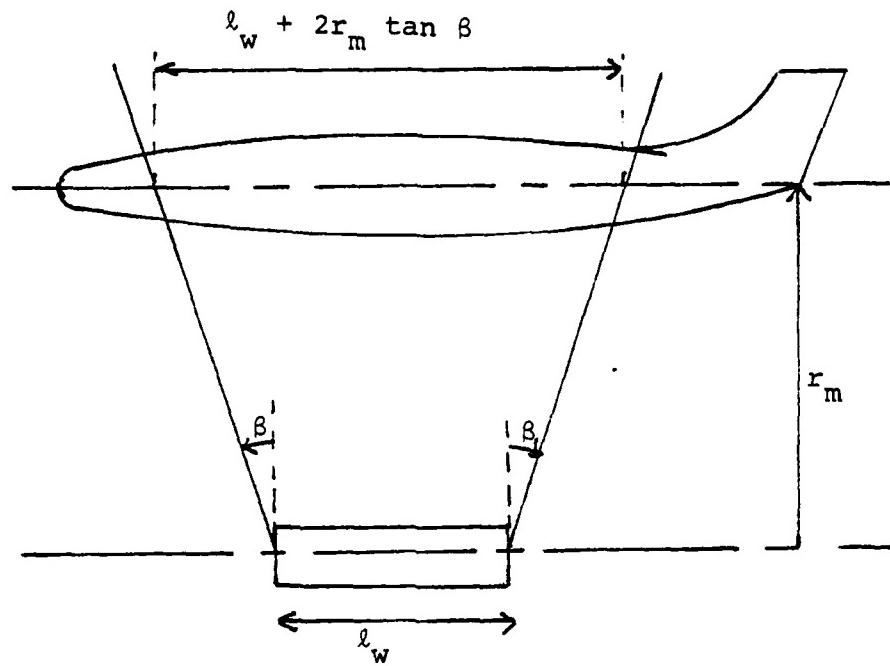


Figure (4-11). Static encounter.

The warhead-target encounter geometry is shown in Figure (4-11). As shown the area of the fragment ring depends on the miss distance, r_m . The area of the ring is given by, A_f .

$$A_f = 2\pi r_m (l_w + 2r_m \tan \beta)$$

If the N fragments are distributed evenly in A_f , the number of fragments per unit area is

$$\rho = \frac{N}{A_f} = \frac{N}{2\pi r_m (l_w + 2r_m \tan \beta)}$$

The average number of hits on a vulnerable component is then,

$$a = \rho A_v \quad 122$$

From equation (9), $A_v = P_{K/H} A_p$

Therefore, $a = \rho P_{K/H} A_p$

$$a = \frac{N P_{K/H} A_p}{2\pi r_m (\ell_w + 2r_m \tan \beta)}$$

If in Figure (4-11), the average target width into the paper is given by \bar{W} , the presented area is,

$$A_p = \bar{W}(\ell_w + 2r_m \tan \beta)$$

The average number of hits is then,

$$a_I = \frac{N P_{K/H} \bar{W}}{2\pi r_m} \quad (11)$$

The distance at which the target just fills the fragment ring is the critical miss distance, and the maximum distance for which a_I applies. This is found by setting

$$L = \ell_w + 2r_c \tan \beta$$

From which

$$r_c = \frac{L - \ell_w}{2 \tan \beta}$$

If the miss distance in Figure (4-11) is such that the entire target is always presented to the fragment ring; i.e., $r_m > r_c$, the presented area becomes,

$$A_p = \bar{W} L$$

The average number of hits then becomes, a_{II}

$$a_{II} = \frac{N P_{K/H} \bar{W} L}{2\pi r_m (\ell_w + 2r_m \tan \beta)} \quad (12)$$

3. Sizing the Warhead Radius

The parameter of interest in this chapter is the diameter of the warhead required to achieve a specified kill probability. The development thus far is the conditional probability of kill, P_D . It has been assumed that the guidance system delivers the warhead to the point of interest, and the fuze detonates the warhead at this point. Since only the conditional probability of kill is determined, the purpose of this section will be to maximize P_D or to find the warhead diameter which sets $P_D = 1$.

From the threat to be encountered, the target presented area, A_p , and the vulnerability, $P_{K/H}$, can be determined. Also from an analysis of the threat, the size and impact velocity of the fragments necessary to kill the target can be determined. The initial velocity required to obtain the impact velocity is a function of the explosive used and the charge to mass ratio, C/M.

$$\frac{C}{M} = \frac{\text{Mass of explosive/unit length}}{\text{Mass of warhead casing/unit length}}$$

The initial velocity is given by Gurney's equation,

$$v_i = \sqrt{2E} \left[\frac{C/M}{1+C/2M} \right]^{1/2} \quad (13)$$

Gurney's constant, $\sqrt{2E}$, for various explosives are given below [11].

Explosive	Density, kg/m ³	$\sqrt{2E}$, m/sec
TNT	1590	2316.5
RDX	1650	2834.6
HMX	1840	3118.1
PETN	1730	2834.6
Tetryl	1620	2500.0
Composition B	1680	2682.2
Octol	1800	2895.6

From equation (13), the charge to mass ratio necessary to attain the specified initial velocity can be determined.

$$\frac{C}{M} = \frac{v_i^2 / 2E}{1 - v_i^2 / 2(2E)}$$

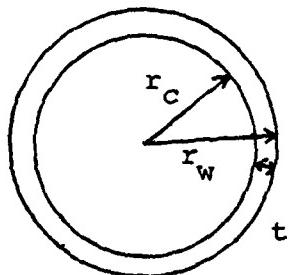


Figure (4-12). Warhead.

From Figure (4-12), C/M can be expressed as,

$$\frac{C}{M} = \frac{\pi r_e^2 \rho_e}{(\pi r_w^2 - \pi r_e^2) \rho_c}$$

where

ρ_e = explosive density

ρ_c = casing density

$$\frac{C}{M} = \frac{r_e^2}{(r_e + t)^2 - r_e^2} \frac{\rho_e}{\rho_c}$$

$$\frac{C}{M} = \frac{r_e^2}{2r_e t + t^2} \frac{\rho_e}{\rho_c}$$

If $t \ll r_e$

$$\frac{C}{M} = \frac{r_e^2}{2r_e t} \frac{\rho_e}{\rho_c} = \frac{\rho_e}{\rho_c} \frac{r_e}{2t}$$

$$\frac{C}{M} = \frac{\rho_e}{\rho_c} \frac{(r_w - t)}{2t} \quad (14)$$

From equation (14) the casing thickness in terms of the warhead radius can be determined.

$$t = \frac{r_w}{2 \frac{C}{M} \frac{\rho_c}{\rho_e} + 1}$$

With the casing thickness fixed as a function of warhead radius the number of fragments, N , can be determined. In order to achieve the desired velocities of fragments, it is essential to have enough length for the diameter. An acceptable length to diameter ratio for a cylindrical warhead is from two to three. Most air-to-air missiles have a length to diameter ratio of 2.5. For this analysis a value of 2.5 is used.

The warhead casing volume is then given by, V_c' ,

$$V_c = 2\pi r_w t l_w$$

$$V_c = 10\pi r_w^2$$

The mass of the casing is, m_c' ,

$$m_c = 10\pi r_w^2 \rho_c$$

The total number of fragments, N , is obtained by dividing the case mass by the individual fragment mass.

$$N = \frac{m_c}{m} = \frac{10\pi r_w^2 \rho_c}{m}$$

From the equation (11) or (12) the average number of hits can be determined as a function of warhead radius

$$a_I = \frac{5\pi r_w^2 \rho_c P_{K/H} \bar{W}}{mr_m}$$

$$a_{II} = \frac{5\pi r_w^2 \rho_c P_{K/H} \bar{W}_L}{mr_m (\ell_w + 2r_m \tan\beta)}$$

The conditional probability of kill can be determined now for a given warhead radius using equation (10).

a. Design Example (Effect of radius on P_D)

If the above equations are programmed for the conditions listed below, a plot of the conditional probability of kill versus warhead radius can be obtained.

Threat: RAM-K

$$L = 19.51 \text{ m}$$

$$\bar{w} = 6.10 \text{ m}$$

$$P_{K/H} = .10$$

Fragments:

$$m = 105 \text{ grains} = 0.0068 \text{ kg}$$

$$v_i = 2133.6 \text{ m/sec}$$

$$\rho_c = 7000 \text{ kg/m}^3$$

$$\beta = 20 \text{ degrees}$$

Explosive: Composition B

$$\rho_e = 1680 \text{ kg/m}^3$$

Miss distance: $r_m = 50 \text{ ft} = 15.24 \text{ m}$

Figure (4-13) is a plot of the output. From this figure it can be seen that a warhead radius of $r_w = .06 \text{ m}$ is required to achieve a conditional kill probability, $P_D = 1.0$. Therefore the missile diameter required for warhead considerations is, $d = 4.72 \text{ inches}$.

The warhead radius also varies with the required initial velocity. From the equations for charge to mass ratio

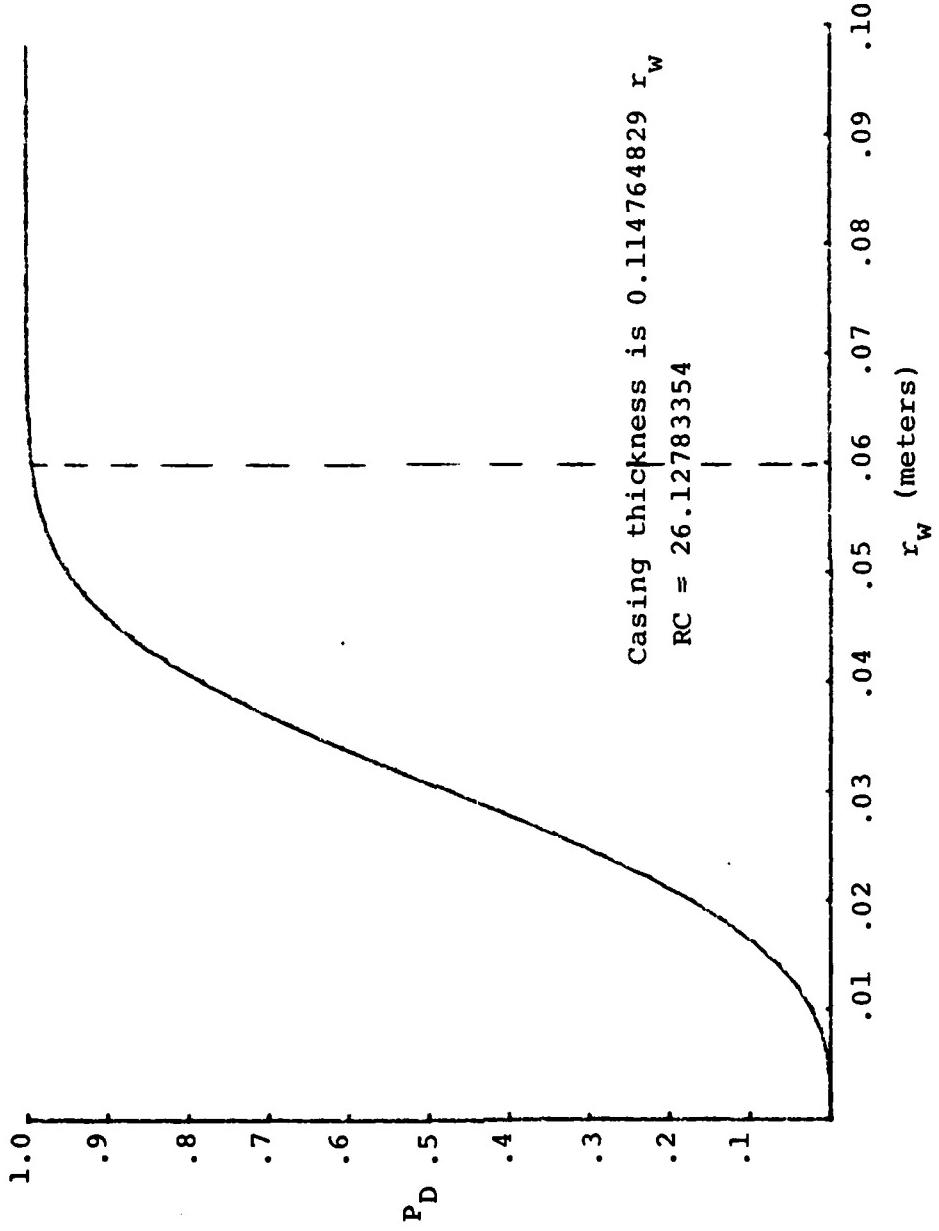


Figure (4-13). P_D versus r_w .

and casing thickness the radius required to achieve a specified kill probability can be determined.

Since,

$$t = \frac{r_w}{2 \frac{C}{M} \frac{\rho_c}{\rho_e} + 1}$$

and

$$\frac{C}{M} = \frac{v_i^2 / 2E}{1 - v_i^2 / 2(2E)}$$

the average number of hits becomes,

$$a_I = \frac{5 \rho_c P_{K/H} \bar{W} r_w^3}{\left[2 \left(\frac{v_i^2 / 2E}{1 - v_i^2 / 2(2E)} \right) \frac{\rho_c}{\rho_e} + 1 \right] m r_m}$$

The above equation assumes $r_m < r_c$

Letting

$$b = \frac{5 \rho_c P_{K/H} \bar{W}}{\left[2 \left(\frac{v_i^2 / 2E}{1 - v_i^2 / 2(2E)} \right) \frac{\rho_c}{\rho_e} + 1 \right] m r_m}$$

the probability of kill becomes,

$$P_D = 1 - e^{-br_w^2} \quad (15)$$

If the conditional probability of kill is selected as $P_D = 0.999$, equation (15) can be solved for the warhead radius.

$$r_w = \left(\frac{6.9078}{b} \right)^{1/3} \quad (16)$$

The initial velocity, v_i , required to achieve a target kill is a function of the miss distance, fragment size, expected encounter altitude and target characteristics. For this simple analysis the effect of initial velocity on warhead radius will be studied. The effect of varying the initial velocity in equation (16) is plotted in Figure (4-14). The results were determined for various explosives to show their effect on the warhead radius. If an initial velocity of 7000 ft/sec is chosen with Composition B as the explosive the required warhead radius is 2.6 inches.

From this analysis the warhead weight can be found. The casing thickness is given by,

$$t = .115 r_w \quad (\text{from required C/M})$$

$$l_w = 5 r_w$$

Therefore, the explosive weight is

$$w_e = \pi(r_w - t)^2 l_w \rho_e$$

$$w_e = 5.9530 \text{ kg}$$

The casing weight becomes,

$$w_c = (\pi r_w^2 - \pi r_e^2) l_w \rho_c$$

$$w_c = 6.8651 \text{ kg}$$

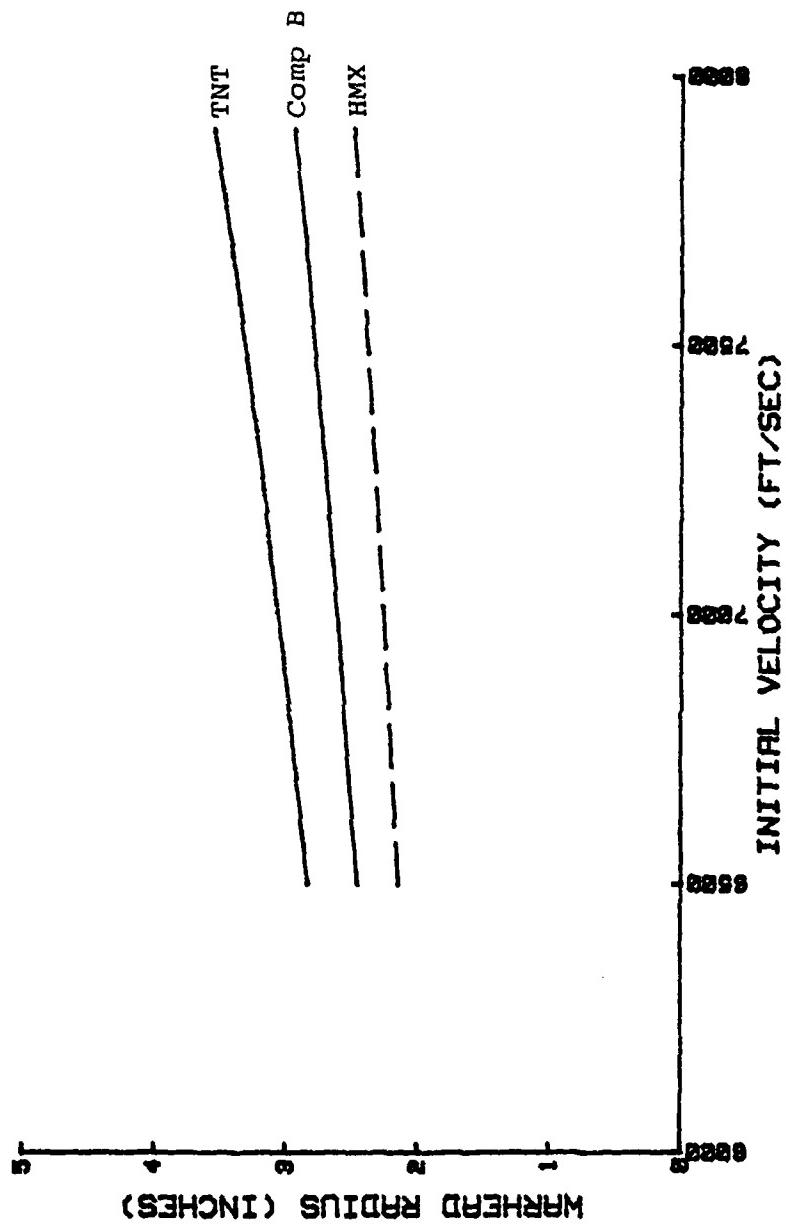


Figure (4-14). Effect of initial velocity on warhead radius.

The total warhead weight becomes,

$$W_{WH} = 12.8181 \text{ kg} = 28.26 \text{ lb}$$

The required diameter for the radar antenna was 10 inches and exceeds that required for the warhead. If the warhead is made hollow and kept at the same weight, equation (14) can be reformulated to give

$$\frac{C}{M} = \frac{[(r_w - t)^2 - r_i^2] \rho_e}{[2r_w t - t^2] \rho_c}$$

The hollow portion of the warhead is then found from

$$\frac{W_e}{\rho_e} = \pi (r_w^2 - r_i^2) \ell_w$$

Figure (4-15) is the resulting warhead.

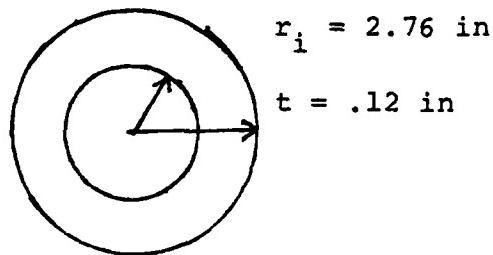


Figure (4-15). Hollow warhead.

V. BASELINE DEFINITION

A. CONTROL CONCEPTS

The lifting and control surfaces of a missile may be of monowing, triwing or cruciform configuration. Figure (5-1) shows these three arrangements.

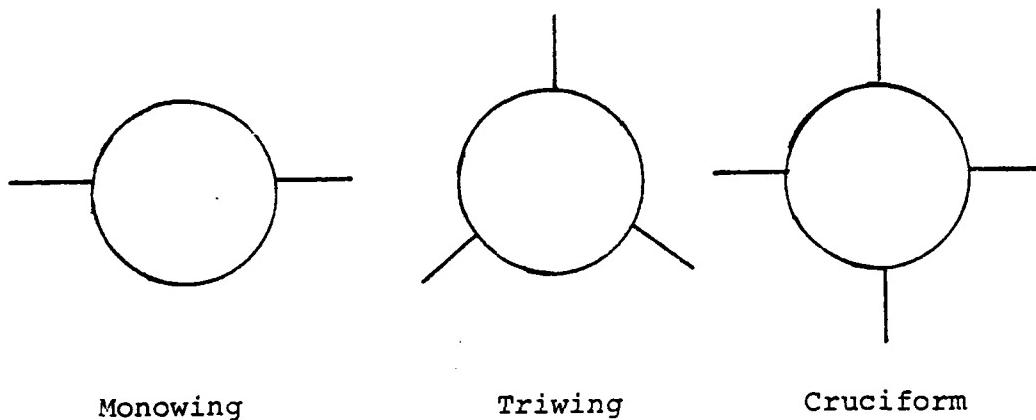


Figure (5-1). Control configurations.

The monowing arrangement is typical of most cruise missiles, which require long range and low drag. For this type of arrangement the missile must bank to orient the lift vector for a maneuver. Because of this, the monowing missile is not as rapid in maneuvering as the cruciform configuration, which can produce lift in any direction instantaneously. The cruciform control also has identical pitch and yaw characteristics which results in a simpler control system. The triwing configuration is used very seldom for conventional missiles. It can be shown that

the triwing missile requires larger wing size; therefore, there is little drag savings even though there is one less wing than with the cruciform missile [10]. From this discussion it can be seen why the cruciform configuration is the most commonly used for tactical missiles of short or medium range.

The positions of the lifting surfaces on the missile body depends on the method of control used for the missile. There are three conventional methods of control for tactical missiles. These are; 1) Wing control; 2) Tail control; 3) Canard control.

1. Wing Control

Wing control missiles normally have large movable wings located slightly behind the missile center of gravity. Because of the small moment arm the wing surface must be relatively large to provide control effectiveness. As would be expected, larger activators are required for moving the wings. A positive deflection of the wings causes a positive normal force; therefore, the missile reacts almost instantaneously; thus making wing control the fastest reacting method of control. Because of the smaller moment arm, the resulting smaller pitching moment and instantaneous lift result in smaller angles of attack. This feature makes wing control missiles attractive for applications where the incidence angle must be kept small. Air breathing applications often use wing control because of inlet performance degradation at higher angles of attack. This type of control is also good for fixed seekers.

2. Canard Control

Canard control is normally the method of control where the movable surfaces are placed as far forward as possible. Because of the resulting larger moment arm, smaller surfaces are required to provide control effectiveness. Lift on the missile is still developed primarily by the aft (wing) lifting surface. Response is slower than wing control because of the need to pitch the missile to an angle of attack before lift is developed. Higher angles of attack are needed to generate the required lift. One advantage of this type of control is convenience of packaging. Since the controls and avionics are forward of the propulsion system, the need for connectors is eliminated. For stability reasons the wing of a canard control missile must be located farther aft than a conventional wing-tail configuration. The zero lift drag is normally lower than wing control missiles due to smaller surfaces.

3. Tail Control

Like the case of canard control, a tail control missile has the movable surface as far from the center of gravity as possible. This also results in a larger moment arm and therefore smaller surface required. Control deflection is the opposite of that for canard or wing control since a negative control deflection results in a pitching moment that pitches the nose up and therefore a positive lift on the main lifting surface (wing). Tail control is normally the slowest method of control. One advantage is that the flight controls are at the end of the

missile requiring the propulsion system to be located forward of other types of control. This results in less center of gravity movement as the propellant grain burns. This becomes a definite advantage for longer range missiles.

B. GROSS WEIGHT AND CENTER OF GRAVITY

Since the component weights and their precise locations cannot be determined at this point in the design process, some method of estimating the total weight and center of gravity is needed. There are two approaches commonly used to find the gross weight of missiles in the conceptual design phase. One is through the use of the historical data discussed in Chapter 2. Since the warhead weight is known, an estimate of the gross weight is now,

$$W_G = \left(\frac{W_G}{W_{WH}} \right)_{AVG} W_{WH}$$

Another method, which is used extensively in the design of aircraft, is the use of regression formulas to find gross weight or component weights in terms of parameters that are known early in the design. Reference (7) has derived such a formula for the gross weight of a missile. It is,

$$W_G = K_G (L)^{2.13} (D)^{1.14}$$

K_G = Constant to be determined (1)

L = Total missile length (inches)

D = Missile diameter (inches)

The constant K_G in equation (1) is derived from a baseline (generic) missile. In this case

$$K_G = \frac{(W_G)_{\text{baseline}}}{(L)^{2.13} (D)^{1.14}}$$

baseline baseline

The accuracy of a regression formula such as equation (1) depends upon how close the synthesized missile is to the baseline for which K_G was determined. If the parameters used vary more than 20 percent, the accuracy of the regression equation decreases rapidly; therefore, if the parameters length and diameter vary significantly from those of the baseline, equation (1) may give an erroneous estimate of the gross weight. As an aid in determining the gross weight, the following values of K_G were derived from data given in reference [7].

Missile	Figure	K_G
SRAAM	(5-2)	.00128
MRAAM	(5-3)	.00118
LRAAM	(5-4)	.00093
SAM	(5-5)	.00108

Figures (5-2), (5-3), (5-4), and (5-5) show the generic missiles from which these values were derived.

The center of gravity of the baseline missile for this section is taken to be 60 percent of the total length. At this point sufficient information has been developed to define the baseline missile from which design iterations will be made. The lifting

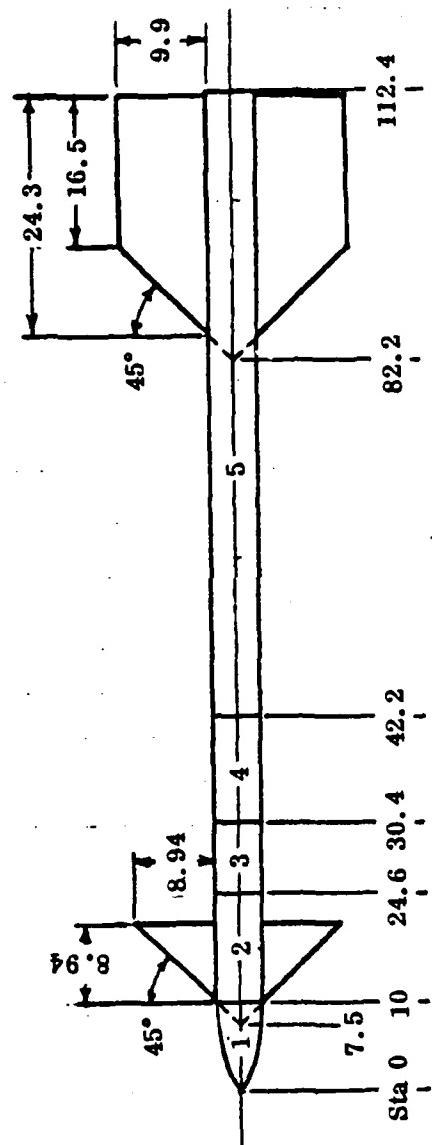
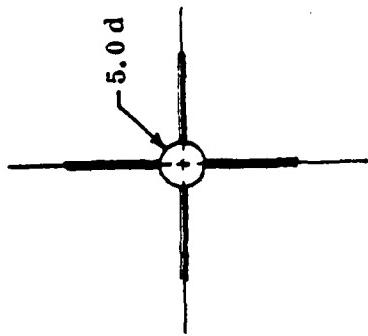


Figure (5-2). Generic SRAAM configuration [7].

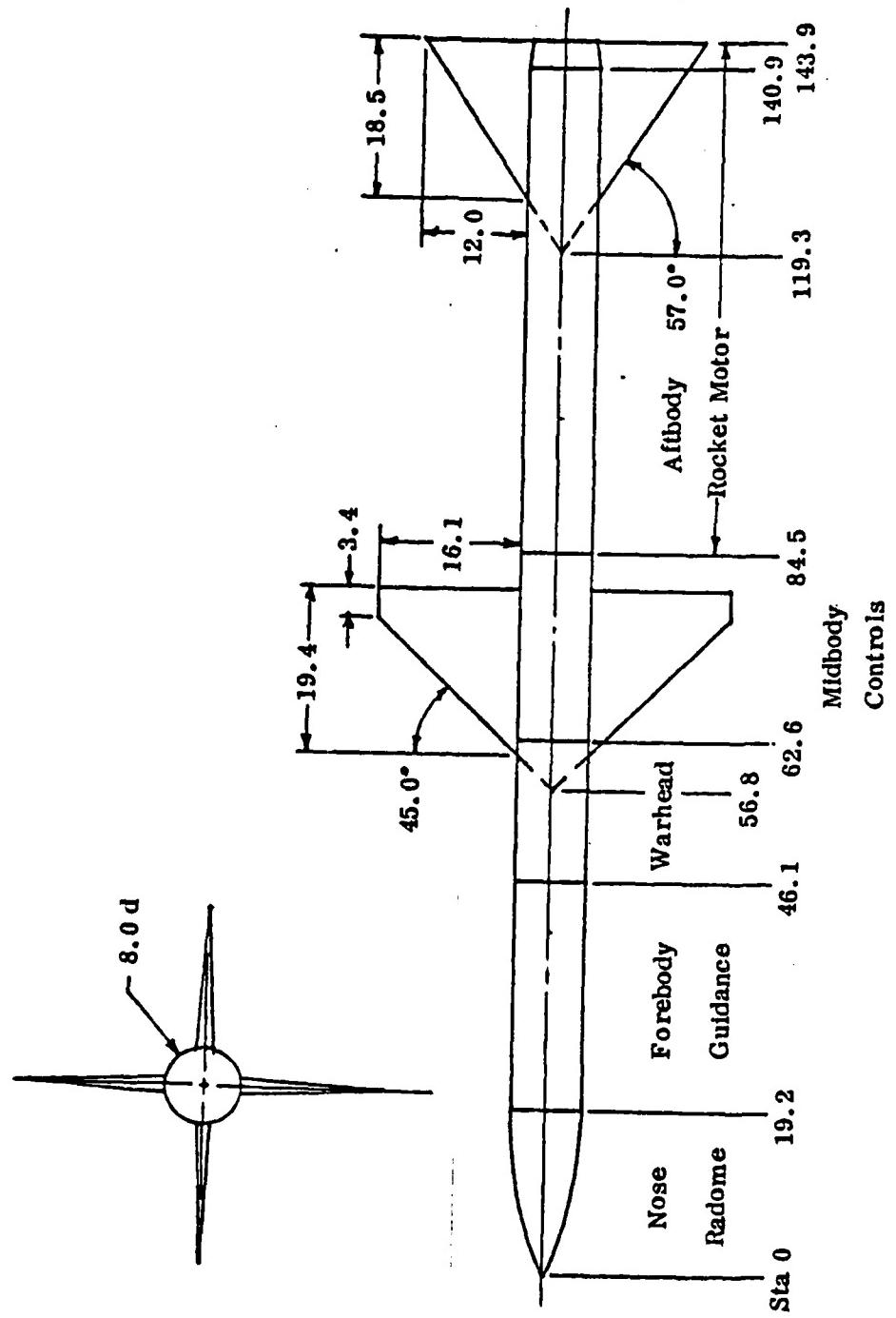


Figure (5-3). Generic MRAAM configuration [7].

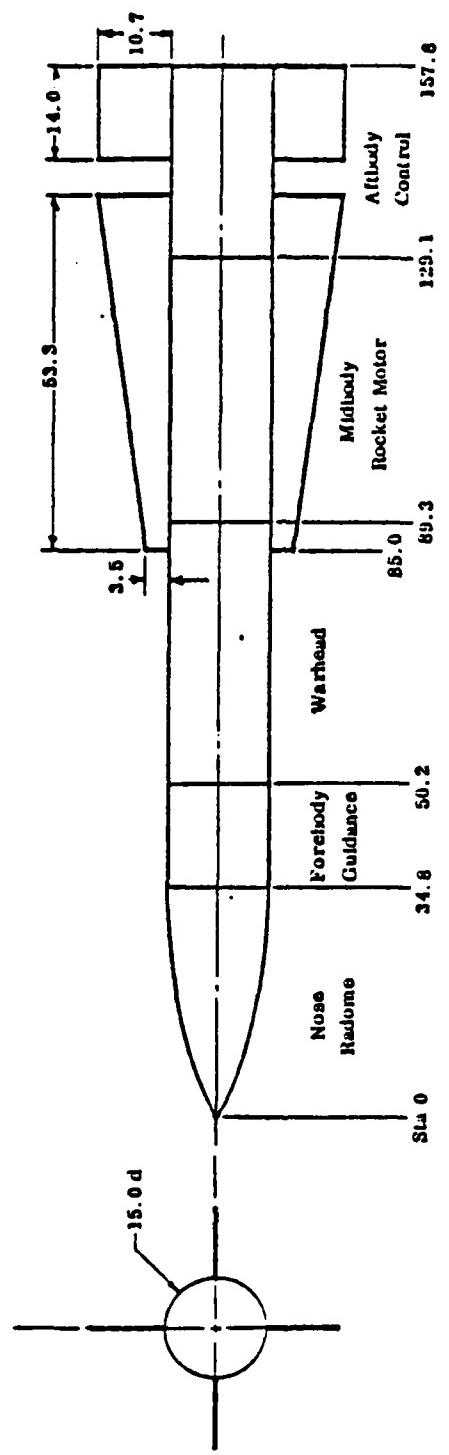


Figure 5-4). Generic LRAM configuration [7].

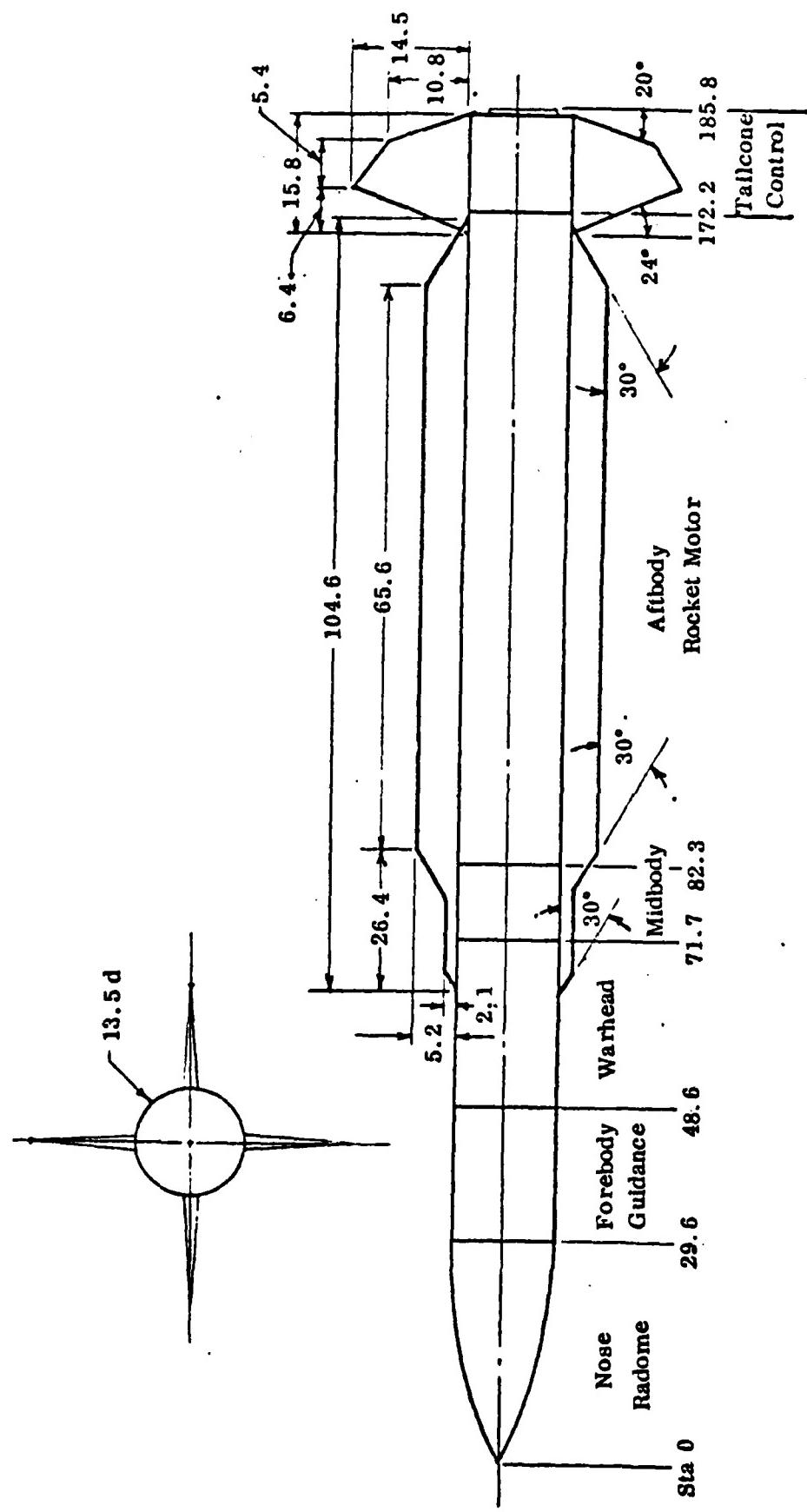


Figure (5-5). Generic SAM configuration [7].

surface planform for the baseline is taken as a delta planform. As will be shown later, this wing planform will be very close to an optimum wing.

C. DESIGN EXAMPLE (BASELINE DEFINITION)

The threat for which this missile is designed is a highly maneuverable fighter. For this threat a canard control, cruciform configuration is chosen. The rationale is that this configuration will provide the fast response necessary at minimum drag. The diameter was fixed at 10 inches due to antenna considerations in Chapter 3. From historical data in Chapter 2,

$$L = \left(\frac{L}{D}\right) D = (15.89) 10 = 158.9 \text{ inches}$$

$$L_N = \left(\frac{L_N}{D}\right) D = 22.3 \text{ inches}$$

The MRAAM of Figure (5-3) is used as a generic missile for selecting K_G . Inserting length and diameter into equation (1), the first estimate of gross weight becomes,

$$W_G = .00118 (158.9)^{2.13} (10)^{1.14} = 794.83 \text{ lb}$$

The total lifting surface required is then, from historical average,

$$S = \left(\frac{S}{W}\right)_{AVG} (W_G) = \left(\frac{1}{88.09}\right) (794.83)$$

$$S = 9.02 \text{ ft}^2$$

Since the canard to wing area ratio is known from historical data,

$$S_c/S_w = 0.20$$

$$S_w = 7.52 \text{ ft}^2$$

$$S_c = S - S_w = 1.50 \text{ ft}^2$$

Lifting surfaces (delta planform). From historical data,

$$AR_w = 1.61$$

$$AR_c = 3.74$$

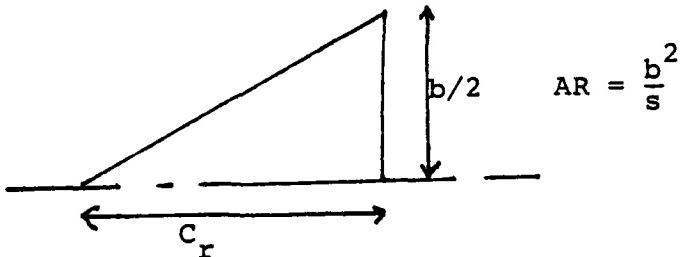


Figure (5-6). Lifting surface.

$$\text{Wing: } (b)_w = \sqrt{AR_w S_w} = 3.48 \text{ ft}$$

$$(c_r)_w = 2 S_w/b = 4.32 \text{ ft}$$

Canard:

$$(b)_c = \sqrt{AR_c S_c} = 2.37 \text{ ft.}$$

$$(c_r)_c = 1.27 \text{ ft.}$$

The baseline missile is now defined. The canards are placed as far forward as possible. The wings are placed as far aft as possible to ensure the center of pressure is behind the center of gravity. The exact location of the wings will be modified in the next chapter. Figure (5-7) is a drawing of the baseline missile.

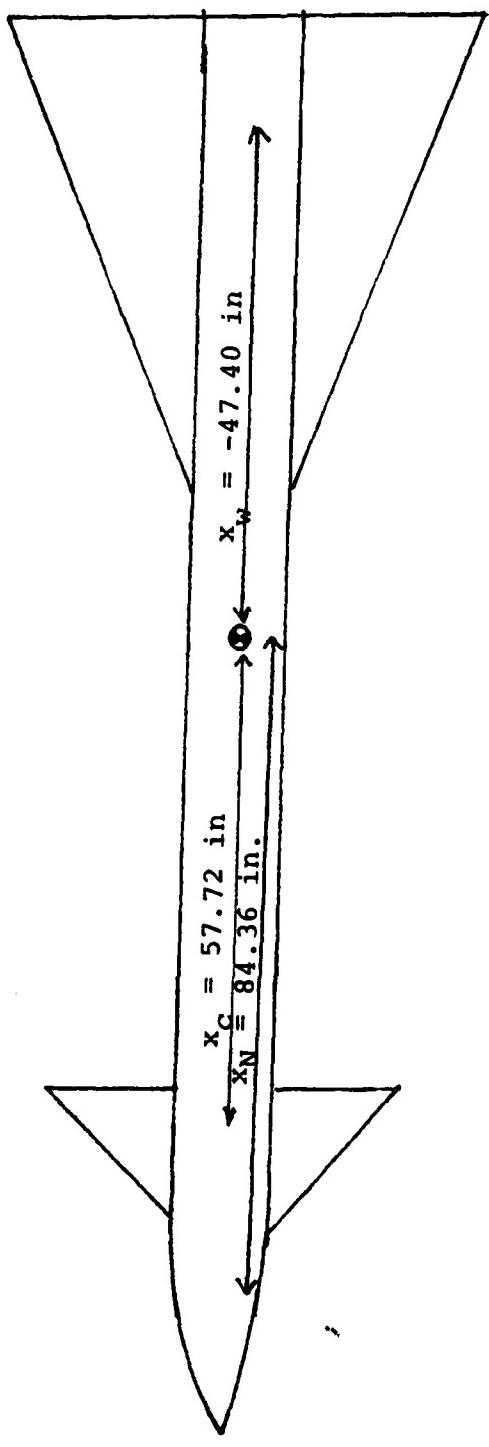


Figure (5-7). Baseline missile.

VI. LINEAR AERODYNAMICS

The total drag on the missile is used to size the propulsion needed for the cruise segment of flight. The wing size depends upon the maximum lift required by the missile. The wing and tail are normally placed to provide minimum drag during cruise or a certain stability margin at launch. To make any of the above calculations, values of the aerodynamic coefficients are needed. This chapter presents the background aerodynamic theory necessary for these initial calculations. The theory used is linear aerodynamic theory and slender body theory, from which simple calculations can be made. Where linear theory did not apply, an attempt was made to find existing empirical expressions, which yield results accurate enough for initial calculations. The full nonlinear theory will be presented in Chapter 9. The reference area for all coefficients in this report is the missile maximum cross sectional area. The reference length is the maximum missile diameter.

A. MISSILE DRAG

The total drag of a missile consists of zero lift drag, C_{D_0} , and induced drag, C_{D_i} .

$$C_D = C_{D_0} + C_{D_i} \quad (1)$$

The zero lift drag can be found from a component build up method in which the contributions due to the nose, body and lifting surfaces are added together to obtain the total zero lift drag.

Care must be taken to reference the appropriate areas when using this method. The total is then multiplied by 1.25 to account for interference effects and variations in skin roughness [12].

$$C_{D_0} = 1.25 \left[(C_{D_0})_N + (C_{D_0})_B + (C_{D_0})_W + (C_{D_0})_T \right] \quad (2)$$

The method used to find the components drag depends upon the speed regime in which the missile is operating. Since this report is concerned primarily with supersonic tactical missiles, supersonic zero lift drag will be discussed here. The component supersonic zero lift drag can be divided into skin friction, C_{D_f} , and wave drag, C_{D_W} .

$$(C_{D_0})_W = (C_{D_f})_W + (C_{D_W})_W$$

1. Supersonic Skin Friction

The flow over a body traveling at supersonic speeds is likely to be turbulent; so the incompressible skin friction coefficient is given by,

$$C_{f_i} = \frac{.455}{(\log_{10} R_e)^{2.58}} \quad (3)$$

The Reynolds number in equation (3) is based upon the cruise altitude and speed and upon the characteristic length for the

component being determined. The Reynolds number is given by.

$$R_e = \frac{\rho V_M x}{\mu}$$

where $x = L_N$, the length of the nose

L_{CB} , the length of the body without the nose

(\bar{c}), the mean aerodynamic chord

The compressibility correction to equation (3) is,

$$C_f = C_{f_i} (1 + 0.15 M_M^2)^{-0.58} \quad (4)$$

From equation (4) the skin friction drag coefficient for each component can be found when referenced to the appropriate area.

$$(C_{D_f})_N = (C_f)_N \frac{(S_{wetted})_N}{S_{ref}}$$

$$(C_{D_f})_B = (C_f)_B \frac{(S_{wetted})_B}{S_{ref}}$$

The lifting surface skin friction drag is determined in a similar manner. Care must be taken to include all surfaces in the wetted area calculation for the lifting surfaces.

2. Supersonic Wave Drag

The supersonic wave drag consists of components contributed by the nose and lifting surface. Nose wave drag depends on the shape of the nose, and the most common nose shapes are

conical, ogival and hemispherical. Reference [10] lists empirical formulas for finding the form(wave) drag of various nose shapes at zero angle of attack.

$$(1) \text{ Conical; } C_{D_W} = (0.083 + 0.097/M_M^2)(\sigma/10)^{1.69}$$

$$\sigma = \tan \frac{D}{2L_N} = \text{Nose semi-vertex angle (Degs)}$$

The center of pressure for a conical nose is at the centroid of the nose planform or two thirds the length of the nose.

$$(2) \text{ Ogival; } C_{D_W} = P \left\{ 1 - \frac{2[196(L_N/D)^2 - 16]}{28(M + 18)(L_N/D)^2} \right\}$$

$$P = (C_{D_W}) \text{ for conical nose}$$

The center of pressure for an ogive noise is,

$$\frac{C_P}{L_N} = \frac{1}{2} \left[\frac{50(M + 18) + 7M^2 P (5M - 18)}{40(M + 18) + 7M^2 P (4M - 3)} \right]$$

The semi-vertix angle for an ogive is twice the equivalent cone angle.

(3) Hemispherical; The drag on a hemispherical nose is extremely high compared to other nose shapes, and is difficult to estimate. An initial estimate of the wave drag can be found from Figure (6-1).

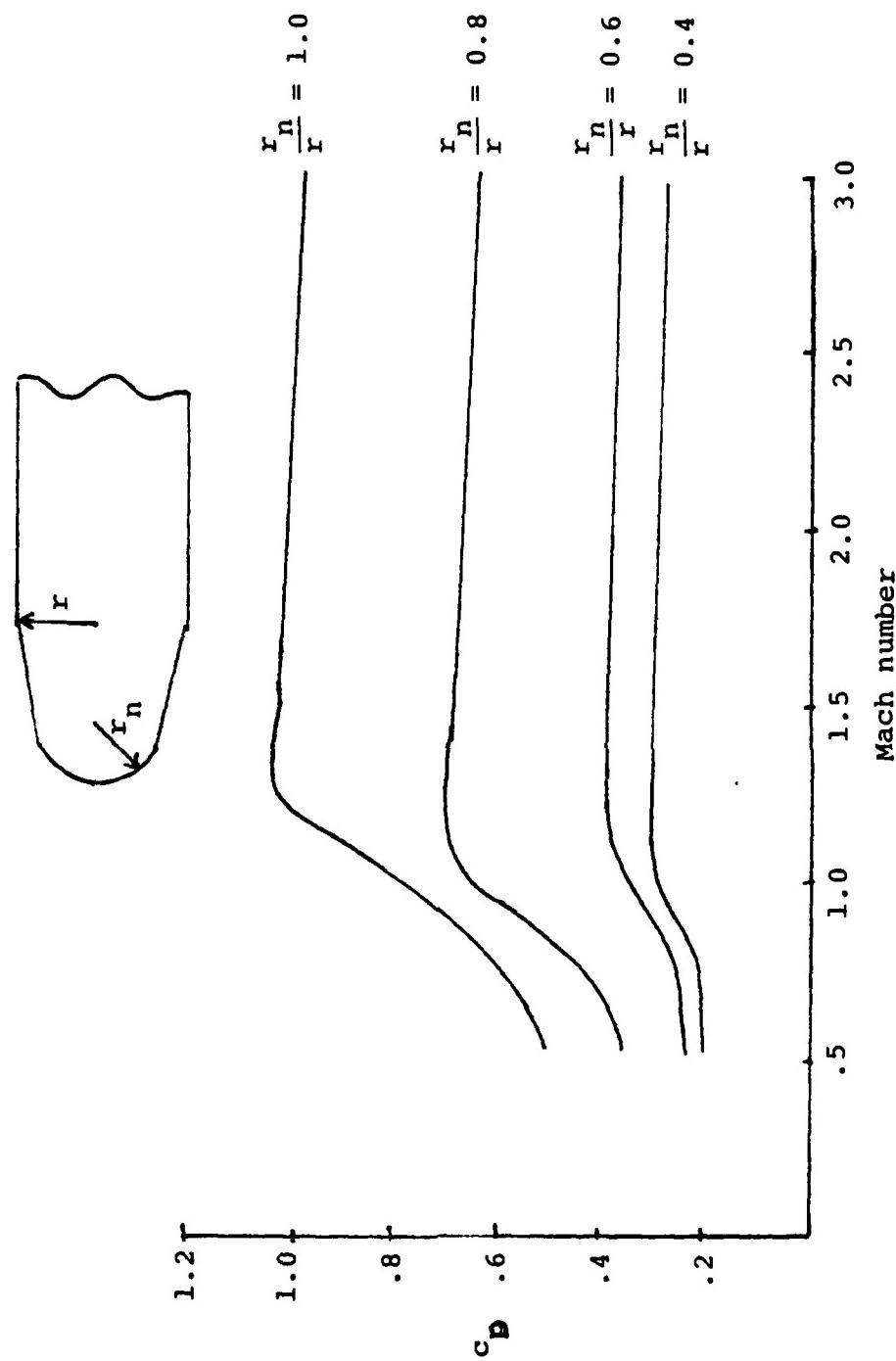


Figure (6-1). Hemispherical nose wave drag.

The wave drag due to the lifting surfaces can be found using the methods of reference [12]. For a double wedge airfoil with sharp leading edges as shown in Figure (6-2), the wave drag is given by the following formulas:

(1) Supersonic leading edge,

$$C_{D_W} = \frac{B}{\beta} \left(\frac{t}{c}\right)^2 \frac{S_W}{S_{ref}}$$

(2) Subsonic leading edge,

$$C_{D_W} = B \cot \Delta_{LE} \left(\frac{t}{c}\right)^2 \frac{S_W}{S_{ref}}$$

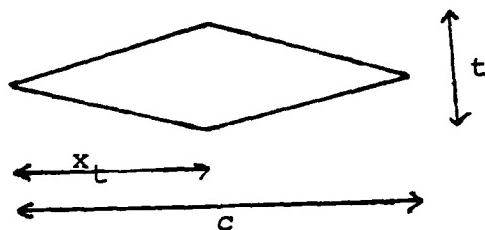


Figure (6-2). Double wedge wing [12].

where $B = \frac{c/x_t}{1 - x_t/c}$ Δ_{LE} = leading edge sweep

 $\beta = \sqrt{M^2 - 1}$ S_W = planform area

3. Base Pressure Drag

The drag contribution of the blunt base for non-boattailed bodies is given in reference [4] as,

$$C'_{AB} = -C_{P_B} = -\frac{2}{\gamma M_M^2} \left\{ \left(\frac{2}{\gamma+1}\right)^{1.14} \left(\frac{1}{M_M}\right)^{2.8} \left[\frac{2\gamma M_M - (-1)}{\gamma+1} \right]^{-1} \right\}$$

This term assumes no jet thrust from the base of the missile, or that the missile is operating in the power off condition. This term is not included in equation (2), which is not a bad assumption for powered flight where the nozzle exit area is approximately equal to the base area of the missile. For the case where the nozzle exit area is much less than the base area as in Figure (3), the base pressure contribution should be included.

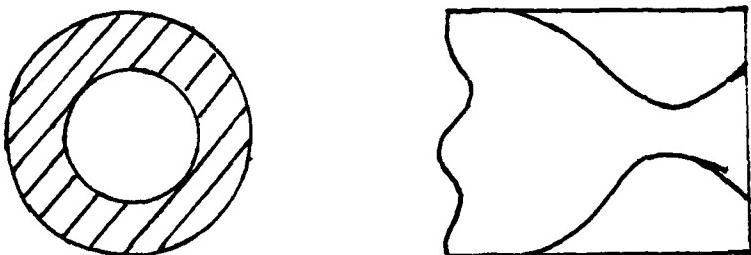


Figure (6-3). Base pressure areas [10].

In this case the base drag is,

$$C_{AB} = C_{AB}' \frac{S_b}{S_{ref}}$$

S_b = shaded area of Figure (6-3)

4. Induced Drag

The induced drag on a missile is the drag due to lift. This drag is caused by the component of the lift vector in the drag direction. For supersonic flow the induced drag is given by,

$$C_{D_i} = \frac{1}{C_{N\alpha}} C_L^2$$

where;

$$C_L = \frac{W}{q S_{ref}}$$

The lift curve slope, $C_{N\alpha}$, will be developed later in this chapter.

B. DESIGN EXAMPLE (ZERO-LIFT DRAG CALCULATION)

The thickness to chord ratio of the wing and tail have not yet been determined; however, it is desirable to construct the lifting surface as thin as structurally possible to minimize the wave drag. Since structures have not been covered the minimum thickness to chord ratio is estimated at 3 percent. The flight and geometric conditions determined thus far are;

$$\begin{aligned}
h &= 10,000 \text{ ft} & L_N &= 1.8583 \\
M_M &= 3.0 & L_{CB} &= 11.3833 \\
S_{ref} &= 0.5454 & (\bar{c})_W &= 2.8816 \\
(t/c)_W &= (t/c)_C = 0.03 & (\bar{c})_C &= 0.8467 \\
S_W &= 7.52 \text{ ft} \\
S_C &= 1.50 \text{ ft}
\end{aligned}$$

From these conditions the following table can be constructed.

	<u>Nose</u>	<u>Afterbody</u>	<u>Wings</u>	<u>Canards</u>
x	1.8583	11.3833	2.8816	0.8467
S_wetted	3.3394	29.8014	15.0400	3.0000
R_e	2.984	18.282	4.6280	1.360
C_{fi}	0.0025	0.0020	0.0024	0.0029
C_f	0.0015	0.0012	0.0015	0.0017
C_{Df}	0.0095	0.0652	0.0827	0.0187
C_{DW}	0.1545	----	0.0400	0.0050
C_{D0}	0.1640	0.0652	0.1227	0.0237

The zero lift drag for the wings and canards in the above table take into account that there are two sets of wings (4 panels). The total missile zero lift drag then becomes,

$$C_{D0} = 1.25 \left[(C_{D0})_N + (C_{D0})_{AB} + (C_{D0})_W + (C_{D0})_C \right]$$

$$C_{D0} = 0.4695$$

C. MISSILE LIFT CURVE SLOPE

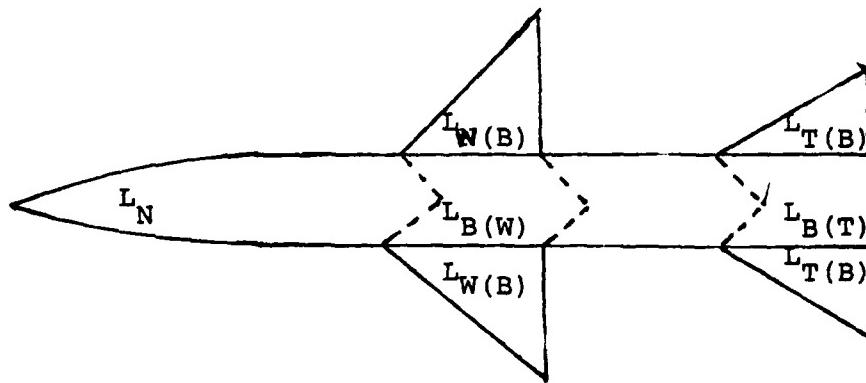


Figure (6-4). Wing-body-tail lift [13].

For the purposes of this section lift on a wing-body-tail combination can be taken as the sum of the components in Figure (6-4). These consist of,

L_N = lift on the nose

$L_W(B)$ = lift on the wing in the presence of the body

$L_B(W)$ = additional lift on the body due to the presence of the wing

$L_T(B)$ = lift on the tail in the presence of the body

$L_B(T)$ = additional lift on the body due to the presence of the tail

The lift of only the wing-body combination can be defined as, L_C' , where,

$$L_C' = K_C L_W \quad (5)$$

The lift of the wing alone, L_W , is that obtained from thin airfoil theory or experiment considering the exposed area only. The constant K_C in equation (5) is defined as

$$K_C \equiv K_{B(W)} + K_{W(B)}$$

From this equation, $K_{B(W)}$ is the ratio of the additional body lift in the presence of the wing to the wing alone lift for zero control deflection, $\delta = 0$.

$$K_{B(W)} = \frac{L_{B(W)}}{L_W} = \frac{(C_{L\alpha})_B(W)}{(C_{L\alpha})_W}, \quad \delta = 0$$

$$K_{W(B)} = \frac{L_{W(B)}}{L_W} = \frac{(C_{L\alpha})_W(B)}{(C_{L\alpha})_W} \quad \delta = 0$$

The interference factors, $K_{B(W)}$ and $K_{W(B)}$ have been determined from slender body theory for wing-body combinations.

Figure (6-5) is a plot of these values. The wing alone lift curve slope, $(C_{L\alpha})_W$ is determined from thin airfoil theory or experiment. The lift curve slope of delta wings with supersonic leading edges is given by,

$$C_{L\alpha} = \frac{4}{\beta} \quad \text{where} \quad \beta = \sqrt{M_M^2 - 1} \quad k < \epsilon$$

For subsonic leading edges this becomes.

$$C_{L\alpha} = \frac{2\pi\tan\epsilon/\tan\mu}{E\beta} \quad k > \epsilon$$

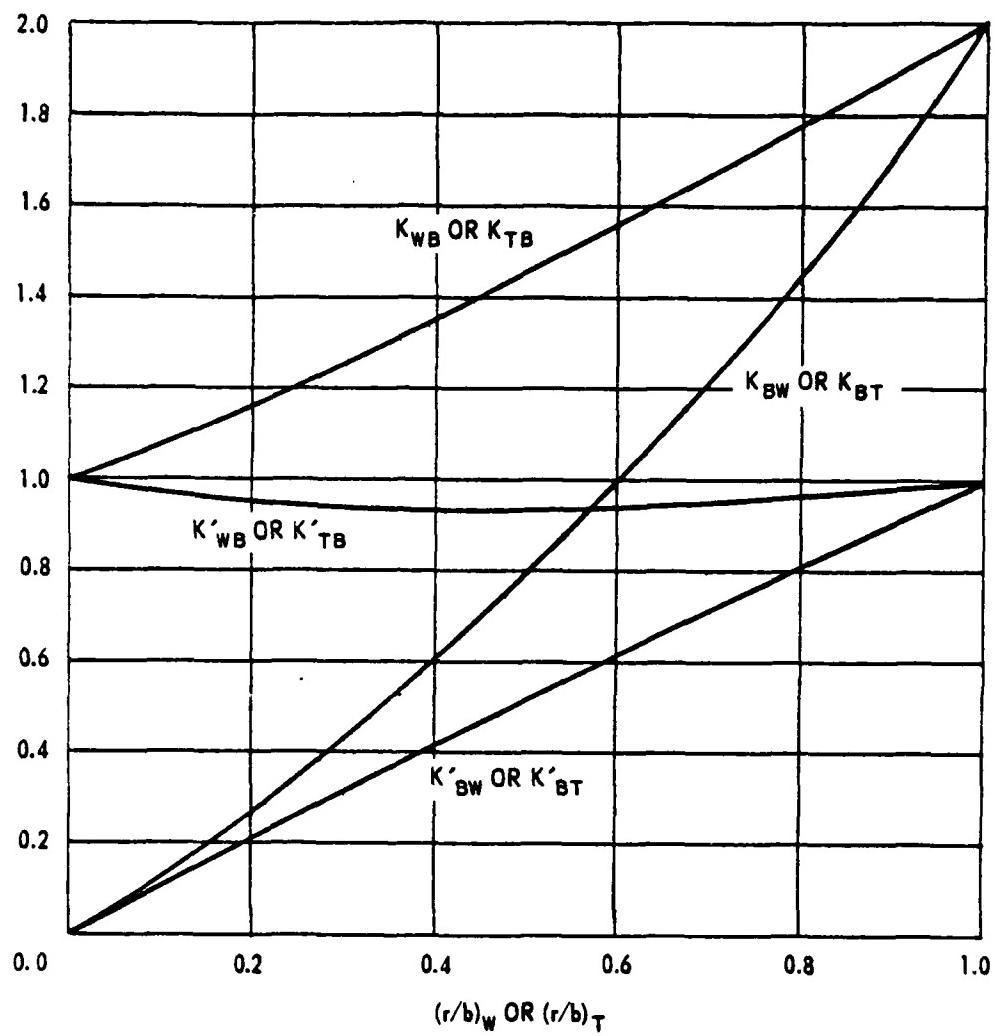
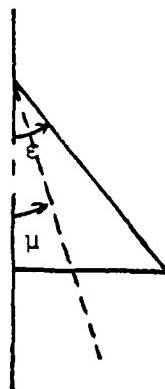


Figure (6-5). Interference factors [12].

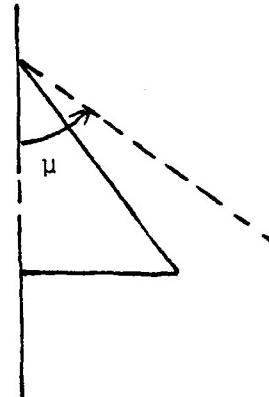
Where E is the elliptic integral of the second kind for

$$\sqrt{1 - (\tan \varepsilon / \tan \mu)},$$

Figure (6) shows the Mach angle μ and sweep angle, ε .



Supersonic LE



Subsonic LE

Figure (6-6). Wing leading edge.

The lift of the nose is that obtained from slender body theory. For small angles of attack

$$(C_{N\alpha})_N = 2/\text{rad.}$$

The tail-body combination lift is determined in the same manner as the wing. The above equations are defined for the case where the incidence angle is zero and the angle of attack, (α),

is varied. Analogous terms can be defined for the case where the angle of attack is zero and the incidence angle (δ) is varied,

$$k_{B(T)} = \frac{L_B(T)}{L_T} = \frac{(C_{L\delta})_B(T)}{(C_{L\alpha})_T} ; \quad \alpha = 0$$

$$k_{T(B)} = \frac{L_T(B)}{L_T} = \frac{(C_{L\delta})_T(B)}{(C_{L\alpha})_T} ; \quad \alpha = 0$$

The interference factors $k_{B(T)}$ and $k_{T(B)}$ were also found from slender body theory and are plotted in Figure (6.5).

From the above definitions the total missile lift curve slope can be found. If the analysis is for small angles of attack so that $L \approx N$.

$$(N)_{cm} = (N)_N + (N)_{W(B)} + (N)_{B(W)} + (N)_{T(B)} + (N)_{B(T)}$$

in coefficient form,

$$\begin{aligned} C_N q S_{reg} &= (C_N)_N q S_{ref} + (C_N)_{W(B)} q S_W + (C_N)_{B(W)} q S_W \\ &\quad + (C_N)_{T(B)} q_T S_T + (C_N)_{B(T)} q_T S_T \end{aligned}$$

If the downwash is neglected so that $q_T = q$, and the above equation is differentiated with respect to angle of attack,

$$\begin{aligned}
 (C_{N\alpha})_{CM} &= (C_{N\alpha})_N + (C_{N\alpha})_{W(B)} \frac{s_w}{s_{ref}} + (C_{N\alpha})_{B(W)} \frac{s_w}{s_{ref}} \\
 &\quad + (C_{N\alpha})_{T(B)} \frac{s_T}{s_{ref}} + (C_{N\alpha})_{B(T)} \frac{s_T}{s_{ref}}
 \end{aligned}$$

From the definition of the interference factors.

$$\begin{aligned}
 (C_{N\alpha})_{CM} &= (C_{N\alpha})_N + (C_{N\alpha})_W (K_{W(B)} + K_{B(W)}) \frac{s_w}{s_{ref}} \\
 &\quad + (C_{N\alpha})_T (K_{T(B)} + K_{B(T)}) \frac{s_T}{s_{ref}} \quad (6)
 \end{aligned}$$

If the tail is the control surface, a similar development for the control effectiveness, $C_{N\delta}$ yields.

$$(C_{N\delta})_{CM} = (C_{N\alpha})_T (k_{T(B)} + k_{B(T)}) \frac{s_T}{s_{ref}} .$$

D. MISSILE PITCHING MOMENT

From Figure (6-7) the moment about the center of gravity can be found. The centroid of the wing is the location of the center of pressure of a wing alone in supersonic flow. The effect of the wing-body combination is to move the center of pressure aft and as can be seen in Figure (6-7) the center of pressure for the additional lift on the body due to the presence of the wing is aft of this location. Reference [13] has an

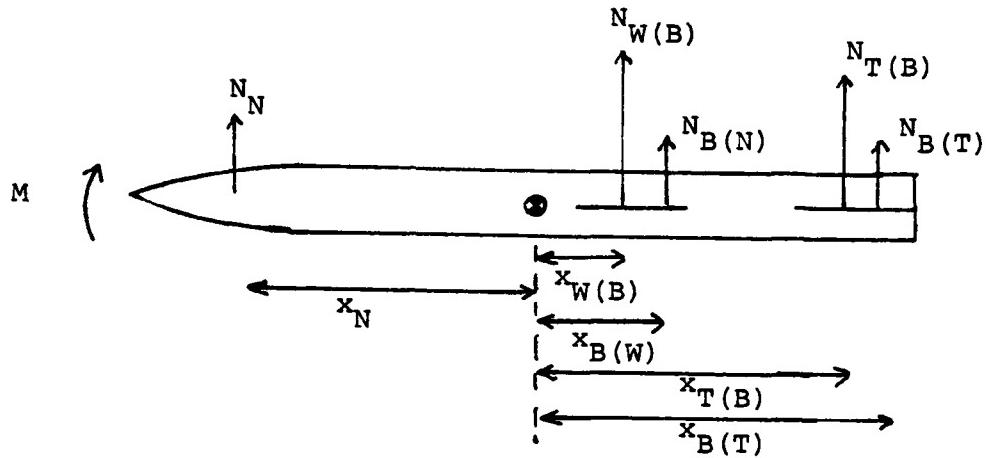


Figure (6-7). Forces Acting on the Missile.

excellent discussion on how to find these centers of pressure. For the purpose of this chapter, which is initial sizing and placement of the lifting surfaces, the center of pressure for both of these forces, $N_{W(B)}$ and $N_{B(W)}$ is taken as the centroid of the wing planform. With this assumption, the moment about the center of gravity is,

$$(M)_{CM} = N_N x_N + (N_{W(B)} + N_{B(W)}) x_W + (N_{T(B)} + N_{B(T)}) x_T$$

Following the same development as for C_N

$$C_{M\alpha} = (C_{N\alpha})_N \frac{x_N}{D} + (C_{N\alpha})_W (K_{W(B)} + K_{B(W)}) \left(\frac{x_W}{D} \right) \left(\frac{S_W}{S_{ref}} \right) \\ + (C_{N\alpha})_T (K_{T(B)} + K_{B(T)}) \left(\frac{x_T}{D} \right) \left(\frac{S_T}{S_{ref}} \right) \quad (7)$$

Also if the tail is the control surface.

$$C_{M\delta} = (C_{N\alpha})_T \left(k_T(B) + k_B(T) \right) \left(\frac{x_T}{D} \right) \left(\frac{S_T}{S_{ref}} \right)$$

One must be careful in defining the moment arms in the above equations. If a nose up pitching moment is developed in Figure (6-7), the moment arm is positive. Conversely, a negative moment arm means a nose down pitching moment is developed.

E. WEIGHT AND CENTER OF GRAVITY VARIATIONS

From the preceding sections of this chapter, it can be seen that the analysis is normally for one point in the flight profile. Since the launch condition has been defined, the tail sizing can be accomplished for this initial condition. The variation of missile weight and center of gravity is due to propellant burning. For a solid propellant this variation can be quite large. Since no information is available on the propellant at this point in the design, the following guidelines will serve for initial calculations. For air-to-air missiles the propellant is approximately 35 percent of the launch weight. The center of gravity travel is 5 percent of the length. For surface-to-air missiles which must be boosted to flight speed, the propellant weight can be taken to be 48 percent of the launch weight with a corresponding center of gravity travel of approximately 8 percent of the length. These are initial approximations taken from historical data and can be refined later in the design process.

VII. LIFTING SURFACE DESIGN

From the analysis of wing lift and drag in Chapter 6, it can be seen that the performance of a wing will vary with planform. Up to this point the lifting surfaces have been considered delta planforms with zero taper ratio. This is not necessarily the optimum planform since it was taken from a historical average and does not apply to a specific missile. This chapter deals with the sizing, placement and planform definition of the missile.

A. WING PLACEMENT

The wing placement on the missile depends upon the type of control used. For canard control the wing (aft surface) is normally fixed as far aft as possible for stability purposes. For a wing control or tail control missile, where the wing is near the center of gravity, the wing placement becomes more critical and depends upon the stability margin required. Since the drag during the cruise segment includes the drag due to lift, one method to minimize drag would be to place the wing such that zero lift is produced on the tail during cruise. The moment about the center of gravity is zero for trimmed flight, thus equation (7) yields, for $(C_N)_T = 0$.

$$x_W = -\frac{(C_{N\alpha})_N x_N S_{ref}}{(C_{N\alpha})_W S_W (K_{W(B)} + K_{B(W)})}$$

B. MANEUVER LOAD FACTOR

Regardless of the type of missile being designed, it will be required to maneuver in order to intercept its intended target. For an air-to-air or surface-to-air missile the level of this maneuver may be quite large. The maneuver load factor required was found in Chapter 3. The maximum maneuver the missile can sustain depends upon the maximum trimmed normal force the missile can develop.

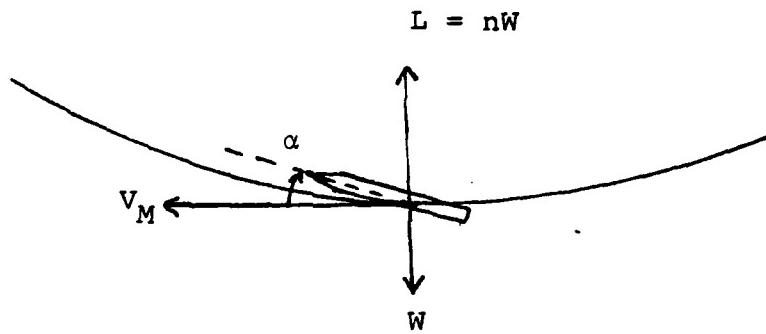


Figure (7-1). Sustained maneuver of a missile.

From Figure (7-1) the force developed by a missile in a constant acceleration turn is

$$L = nW \quad (1)$$

$n = \text{maneuver load factor}$

Small angles of attack are assumed for which the lift and normal force are approximately equal. This approximation is good for angles of attack of up to 10 degrees, and above this value of maximum trimmed angle of attack the linear theory becomes inaccurate. With this assumption for now, equation (1) becomes,

$$L = N = nW$$

$$(C_{N\alpha}) \alpha q S_{ref} = nW \quad (2)$$

$$(C_{N\alpha})_{\text{required}} = \frac{nW}{(\alpha)_{\max} q S_{ref}}$$

Equation (2) gives the lift-curve slope required to develop the required normal force at a trimmed angle of attack, $(\alpha)_{\max}$. The lift curve slope developed is given by equation (6) in Chapter 6.

$$\begin{aligned} (C_{N\alpha})_{CM} &= (C_{N\alpha})_N + (C_{N\alpha})_W (K_W(B) + K_B(W)) \frac{S_W}{S_{ref}} \\ &\quad + (C_{N\alpha})_T (K_T(B) + K_B(T)) \frac{S_W}{S_{ref}} \end{aligned} \quad (3)$$

As discussed earlier the values of the interference factors depend upon the wing planform. Typical values of $(K_W(B) + K_B(W))$ and $(K_T(B) + K_B(T))$ are from 1.5 to 2.0. As a conservative estimate a value of 1.5 is assumed. Equation (3) then becomes,

$$(C_{N\alpha})_{CM} = (C_{N\alpha})_N + 1.5 (C_{N\alpha})_W \frac{S_W}{S_{ref}} + 1.5 (C_{N\alpha})_T \frac{S_T}{S_{ref}} \quad (4)$$

Selecting supersonic leading edges,

$$(C_{N\alpha})_W = (C_{N\alpha})_T = 4/\beta$$

Therefore

$$(C_{N\alpha})_{CM} = (C_{N\alpha})_N + \frac{6}{\beta S_{ref}} (S_W + S_T)$$

If the complete missile lift-curve slope is set equal to the required lift-curve slope, the lifting surface area required to maintain the maneuver load factor can be found, using equation (4).

$$(S_W + S_T)_{req} = [(C_{N\alpha})_{req} - (C_{N\alpha})_N] \frac{\beta S_{ref}}{6}$$

Let

$$K = \frac{(S_W + S_T)_{req}}{(S_W + S_T)_{baseline}}$$

If the same ratio of wing to control surface area is kept to minimize stability perturbations, the new wing and tail area are,

$$S_W = K(S_W)_{baseline}$$

$$S_T = K(S_T)_{baseline}$$

The above analysis assumes a linear variation of $C_{N\alpha}$ which is good for small angles of attack. A more precise analysis will be performed in Chapter 9. The above analysis should be performed at the expected encounter conditions.

1. Design Example (Maneuver Load Factor)

Since a canard control was chosen for the design example the canard and wing position are fixed as far forward and aft as possible. The analysis is done for a conservative Mach number of 2.5 after the missile has slowed from its cruise velocity due to the maneuver. From previous results,

$$n = 31 \text{ g's}$$

$$(S_W + S_C)_{\text{baseline}} = 9.02 \text{ ft}^2$$

$$W_G = 794.83 \text{ lbs}$$

If the missile is required to maneuver at one-half its powered range, and the propellant weight is 35 percent of the launch weight, the maneuver weight becomes

$$W = 655.73 \text{ lbs}$$

The dynamic pressure for $M_M = 2.5$ and at 10,000 feet altitude is

$$q = \frac{1}{2} \rho V_M^2$$

$$q = 6369.84 \text{ lbs ft}^2$$

The required lift-curve slope becomes for $(\alpha_{\max}) = 10 \text{ degrees} = .1745 \text{ rad.}$

$$(C_{Na})_{\text{req}} = 33.5311 \text{ per rad.}$$

The required lifting surface area is;

$$(S_W + S_C)_{\text{req}} = [33.5311 - 2] \frac{\beta S_{\text{ref}}}{6} = 6.57 \text{ ft}^2$$

Therefore $K = 0.7284$.

The required wing and canard sizes to achieve the maneuver are;

$$S_W = (7.52)(.7284) = 5.48 \text{ ft}^2$$
$$S_C = (1.50)(.7284) = 1.09 \text{ ft}^2$$

C. TAIL SIZING

For a wing-tail combination, the primary concern in the sizing of the tail is the static stability of the missile. The missile becomes more stable as the mission proceeds, since the center of gravity moves forward as the propellant burns. As the missile becomes more stable, control of the missile becomes sensitive. If the tail is sized for zero static stability at launch, the missile control will remain more effective during flight. This is the best condition possible without the use of some form of stability augmentation system at launch. Therefore, at launch $C_{Na} = 0$.

$$0 = (C_{Na})_N x_N + (C_{Na})_W x_W \frac{S_W}{S_{ref}} (K_{W(B)} + K_{B(W)})$$
$$+ (C_{Na})_T x_T \frac{S_T}{S_{ref}} (K_{T(B)} + K_{B(T)}) \quad (5)$$

With the lifting surface area fixed due to the maneuver load factor, the tail can be sized to satisfy equation (5). As can be seen the position depends highly on the moment arms that the lift forces act through. Since the missile length has not been fixed at this point in the design and may vary due to propulsion requirements, this analysis will be completed later.

D. WING PLANFORM

The wing planform is specified by the leading edge sweep, Δ_{LE} , taper ratio, λ , aspect ratio, AR and planform area S_W . The planform area was fixed due to maneuvering requirements in a previous section. This section is concerned with defining the remaining planform parameters. Figure (7-2) is the wing planform and the equations used to define these parameters.

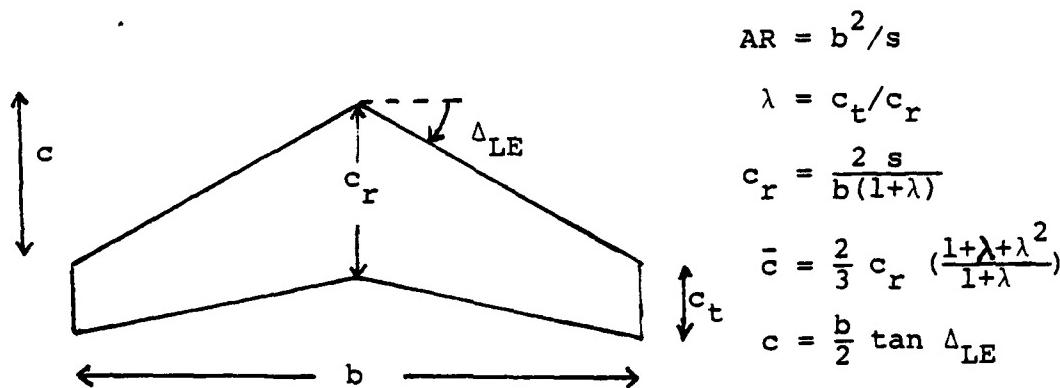


Figure (7-2). Planform geometry.

The value of the lift-curve slope used previously was derived from linear theory and is applicable only to simple planforms. For the purpose of this section, which will include low aspect ratio wings with supersonic and subsonic leading edges, Figures (7-3) and (7-4) are used. These figures have been corrected for 3-D effects.

1. Effect of Taper Ratio and Leading Edge Sweep

The lift and drag characteristics of the wing are the primary parameters of interest. The objective inwing design is

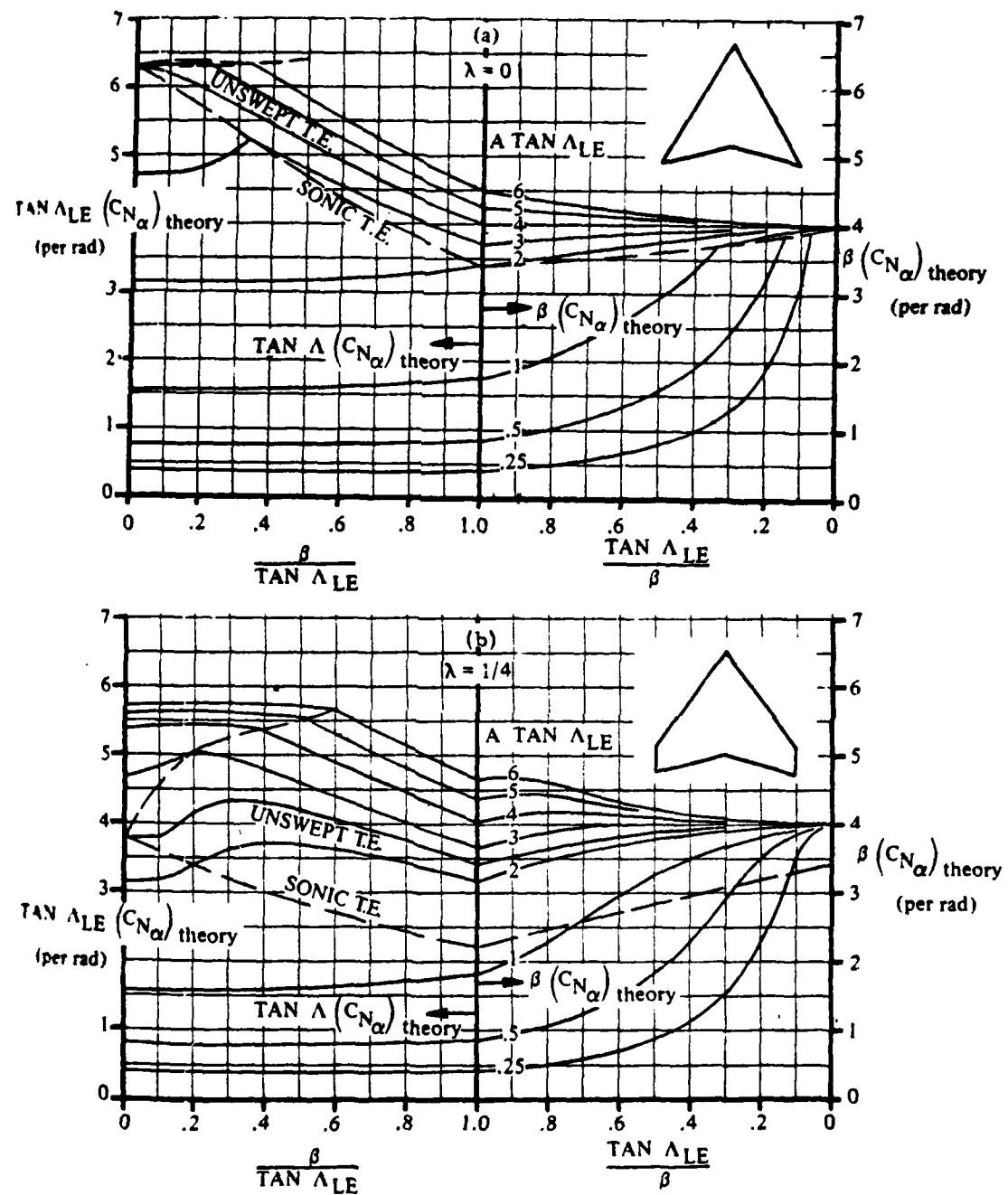


Figure (7-3). Theoretical wing lift curve slope [12].

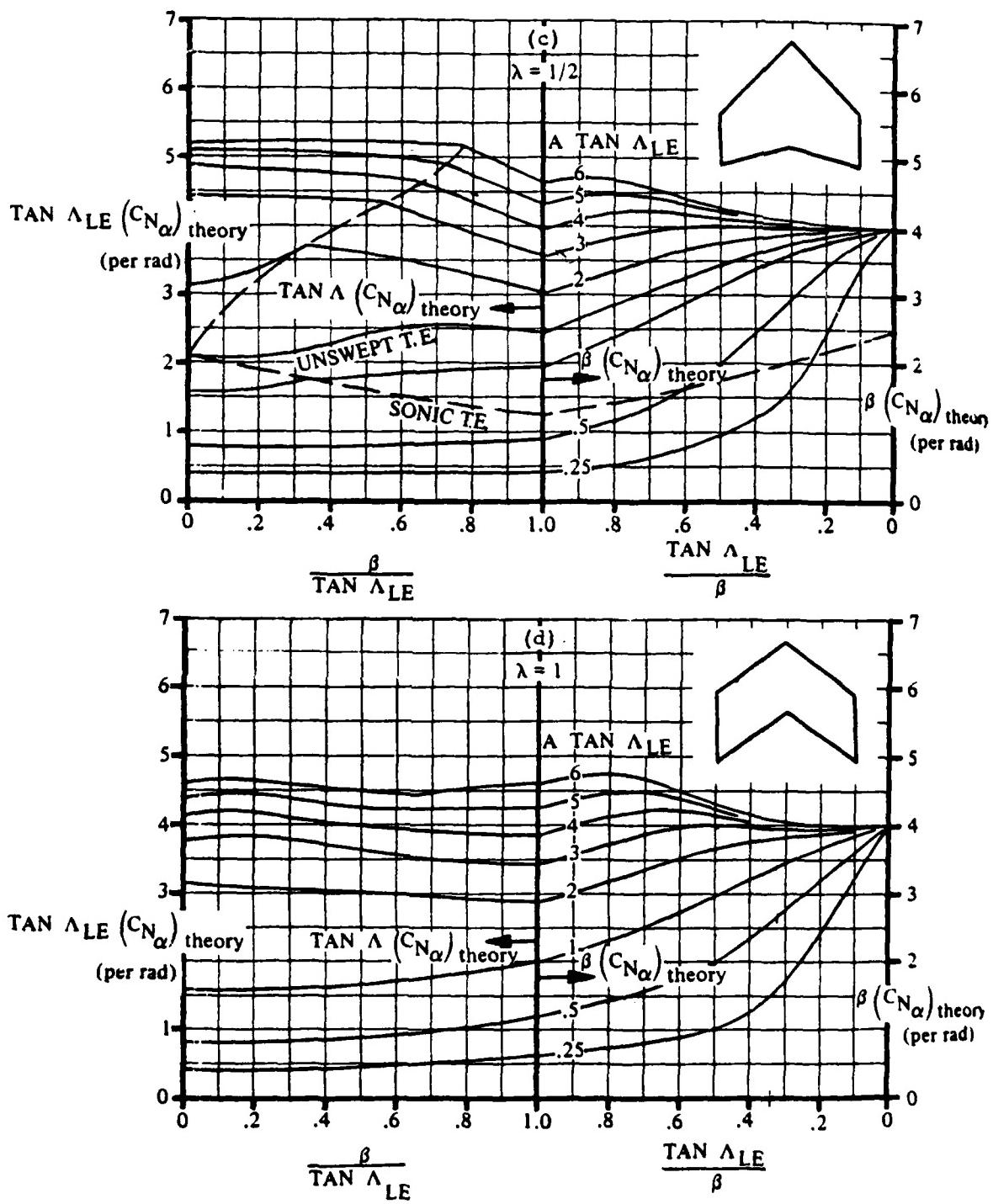


Figure (7-4). Lift curve slope continued [12].

is to obtain maximum lift with minimum drag. It will be shown that these objectives are usually conflicting; therefore, some compromise, or optimum, planform must be found. From Chapter 6 the equations for the drag of a wing in supersonic flow are found. The skin friction drag depends upon the mean geometric chord. Equation (3) in Chapter 6 indicates a larger mean geometric chord would result in less skin friction drag for a fixed area. The mean geometric chord can be expressed in terms of taper ratio as follows,

$$\bar{c} = \frac{2}{3} \frac{2s}{b(1+\lambda)} \left(\frac{1+\lambda+\lambda^2}{1+\lambda} \right)$$

$$\bar{c} = \frac{4}{3} \sqrt{\frac{s}{AR}} \left[\frac{1+\lambda+\lambda^2}{(1+\lambda)^2} \right]$$

This equation leads to a zero taper ratio to maximize \bar{c} and reduce skin friction if the surface area and aspect ratio were fixed. The wave drag is constant for supersonic leading edges, and decreases when the leading edge goes subsonic or the leading edge is behind the mach line. The lift capabilities also decrease as the leading edge goes subsonic.

As stated earlier this leads to conflicting performance since the objective is to minimize drag while maximizing lift. At this point an example best illustrates the results of varying taper ratio and leading edge sweep. The lift - curve slope is derived using the methods of reference (1). Figures (7-3) and (7-4) are from reference [12] and are used to find the supersonic linear lift corrected for 3-D flow effects. The drag methods of Chapter 5 are used to determine the drag characteristics.

2. Example

For a fixed wing area of 4 ft^2 and an aspect ratio of 2 the lift curve slope and drag were determined for a flight Mach number of $M = 2$. This Mach number corresponds to a sonic leading edge sweep of 60° . The wing leading edge sweep was varied from zero to 75° for taper ratios of 0, $1/2$ and 1. The results are plotted in Figures (7-5) and (7-6).

From this example certain generalizations can be made. From Figure (7-5) it can be seen that the drag is relatively insensitive to taper ratio. There is a reduction in drag for increased leading edge sweep. For this example there is approximately a 6 percent drag reduction for every five degree increase beyond the sonic value. Figure (7-6) indicates that the effect of decreasing the taper ratio is to delay the drop in the lift curve slope of the wing. From this example a general guideline would be to fix the leading edge sweep 5 degrees beyond the sonic value and the taper ratio at zero. This would provide a 6 percent reduction of wave drag while maintaining the maximum lifting capabilities of the wing.

3. Effect of Varying Aspect Ratio

The result of increasing aspect ratio is an increase in the lift-curve slope of the wing [12]. The aspect ratio is given by,

$$AR = b^2/s$$

For a zero taper ratio wing this becomes;

$$AR = 2b/c_r \quad (6)$$

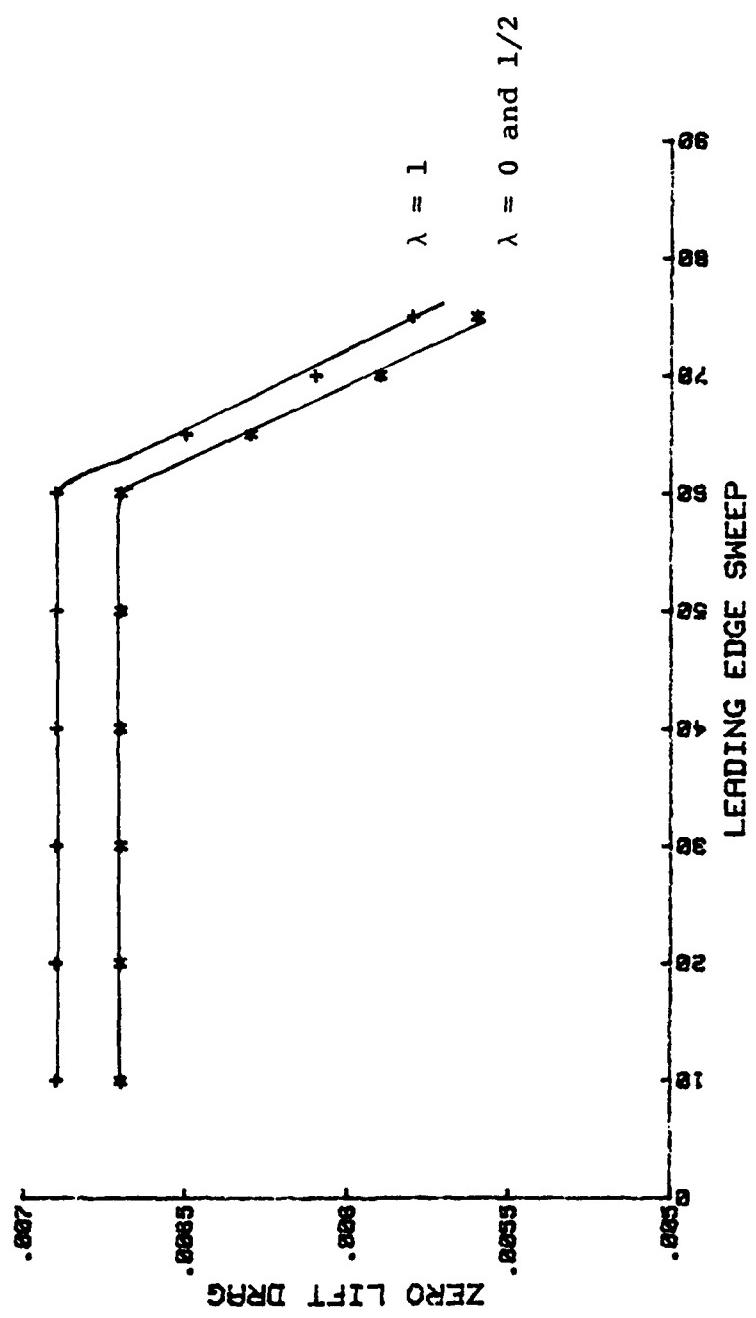


Figure (7-5). Effect of λ on drag.

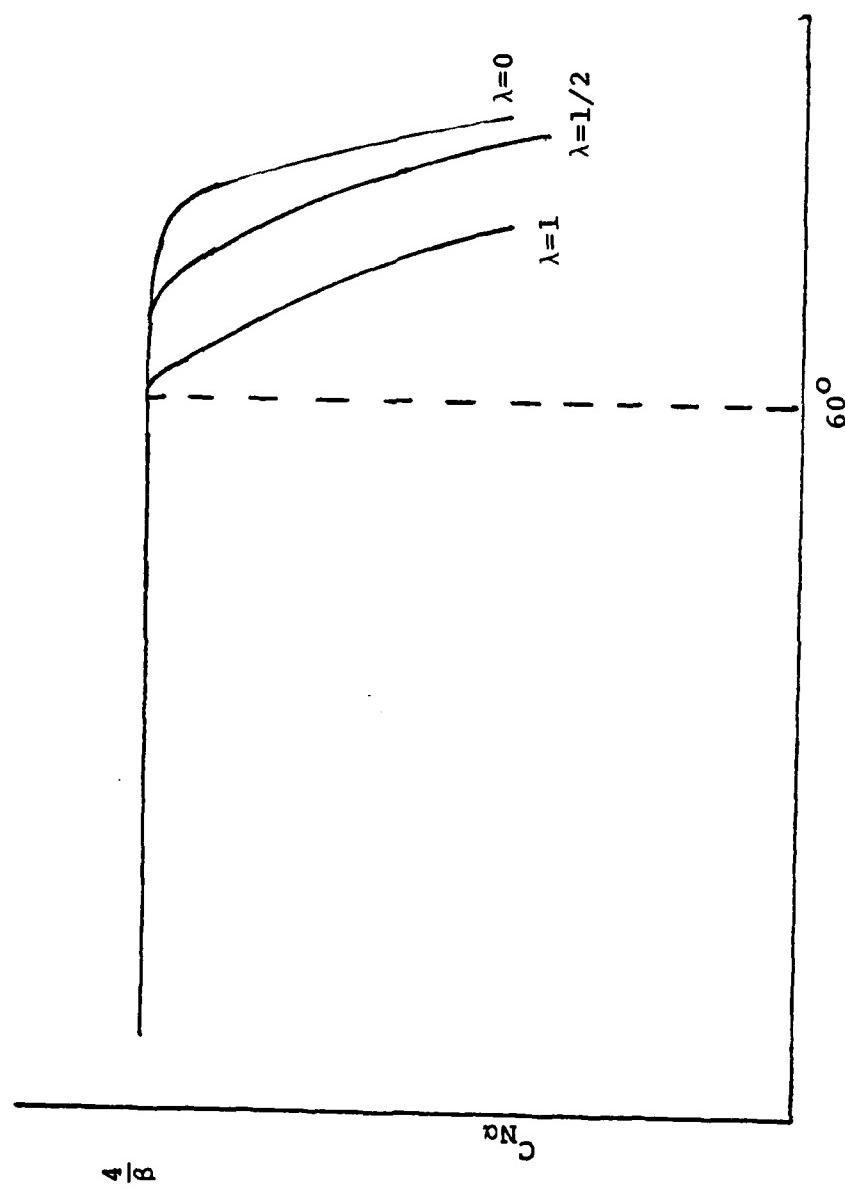


Figure (7-6). Effect of λ on lift curve slope.

From equation (6) it can be seen that increasing aspect ratio results in a corresponding increase in wing span for a given root chord. Missiles are normally span limited, due to launcher constraints; therefore, there is a maximum span which can be accommodated. The drag of the wing again is in conflict with the lift since increasing aspect ratio decreases the mean geometric chord for a given span and therefore increases skin friction drag. A compromise wing AR must be found which considers both the lift and the drag characteristics. For a discussion of optimum aspect ratio, the following functions are defined for convenience:

$$F = F_1 + F_2 \text{ = Lift-drag function} \quad (7)$$

where

$$F_1 = \frac{C_{D0}}{(C_{D0})_{\max}} \text{ = Normalized drag function}$$

$$F_2 = \frac{1/C_{Na}}{1/(C_{Na})_{\min}} \text{ = Normalized lift function}$$

From equation (7) if F is plotted over the allowable range of aspect ratio, a minimum value of the lift-drag function fixes the desired aspect ratio. This aspect ratio corresponds to minimizing the drag function, F_1 , while maximizing the lift function, F_2 . For convenience the abscissa is plotted as $\bar{c}/\bar{c}_{(\max)}$ as shown in Figure (7-7).

From Figure (7-7) the optimum mean geometric chord corresponds to point A.

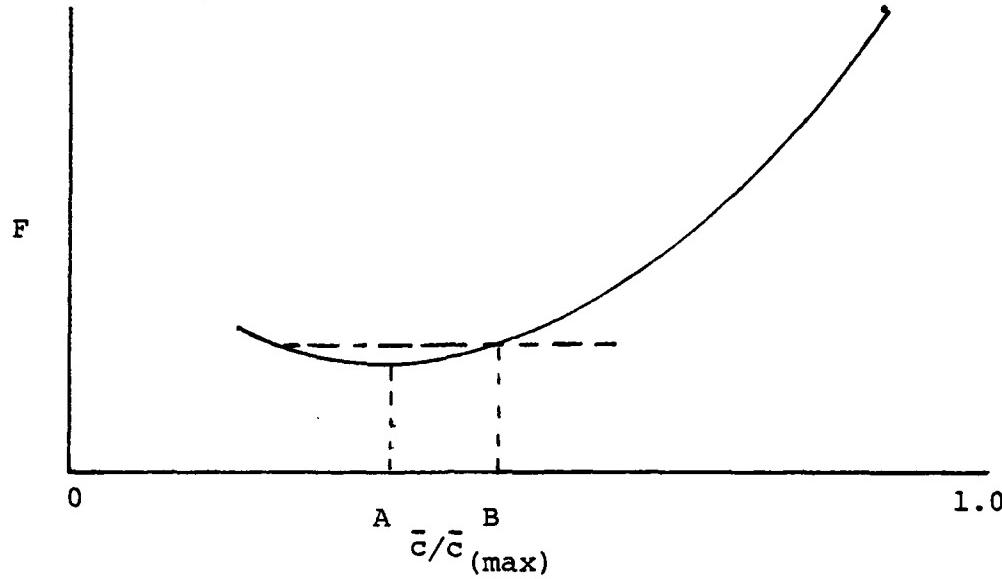


Figure (7-7). Lift-drag function.

$$\bar{c} = \left(\frac{\bar{c}}{\bar{c}_{\max}} \right)_A \bar{c}_{\max}$$

The aspect ratio is then given by

$$c_r = \frac{3}{2} \bar{c}$$

$$b = 2s/c_r$$

$$AR = b^2/s$$

As mentioned earlier a missile is normally span limited. The plot of the lift-drag function F is normally fairly flat on the left, or for $\bar{c}/\bar{c}_{(\max)}$ approaching zero. For this reason point B may be chosen as the optimum since F varies very little up

to this region. Point B corresponds to increased chord and decreased span.

4. Design Example (Wing Planform)

From the previous analysis the following parameters were defined.

$$M_M = 3.0$$

$$S_W = 5.48 \text{ ft}^2$$

If the leading edge sweep is fixed 5 degrees behind the mach line.

$$\mu = \sin^{-1} \frac{1}{M} = 19.5^\circ$$

$$\Delta_{LE} = 95 - \mu = 75.5^\circ$$

$$\lambda = 0$$

The planform table becomes

<u>AR</u>	<u>b</u>	<u>c_r</u>	<u>\bar{c}</u>	<u>$K_{W(B)}$</u>	<u>$K_{B(W)}$</u>
2.0	3.31	3.31	2.21	1.17	.26
1.5	2.87	3.82	2.55	1.19	.30
1.0	2.34	4.68	3.12	1.22	.35
.75	2.03	5.40	3.60	1.24	.40
.50	1.66	6.62	4.41	1.29	.48
.25	1.17	9.36	6.24	1.36	.62

The wave drag is constant and is given by,

$$C_{D_W} = 4 \cot \Delta_{LE} \left(\frac{t}{c}\right)^2 \frac{S_W}{S_{ref}}$$

$$C_{D_W} = .0094$$

AR	C_{D_f}	C_{D_0}	AR $\tan \Delta_{LE}$	$\frac{\beta}{\tan \Delta_{LE}}$	$(C_{Na})_W (K_W(B) + K_B(W))$
2.0	.0303	.0397	7.73	.73	2.1137
1.5	.0296	.0390	5.80	.73	1.9845
1.0	.0288	.0382	3.87	.53	1.8271
.75	.0282	.0376	2.90	.73	1.7813
.50	.0274	.0368	1.93	.73	1.4809
.25	.0260	.0354	.97	.73	.7937

AR	\bar{c}/\bar{c}_{max}	F_1	F_2	F
2.0	.35	1.0000	.3755	1.3755
1.5	.41	.9824	.3999	1.3823
1.0	.50	.9622	.4344	1.3966
.75	.58	.9471	.4456	1.3927
.50	.70	.9270	.5360	1.4630
.25	1.00	.8917	1.0000	1.8917

From Figure (7-8) it can be seen that the lift-drag function remains relatively constant up to $\bar{c}/\bar{c}_{max} = .58$. Since the object is to make the span as small as possible this point is taken. The wing planform becomes,

$$\bar{c} = .58 \bar{c}_{\max} = 3.62 \text{ ft} = 43.44 \text{ in}$$

$$c_r = 5.43 \text{ ft} = 65.16 \text{ in}$$

$$b = 2.02 \text{ ft} = 24.24 \text{ in}$$

$$c = 3.90 \text{ ft} = 46.80 \text{ in}$$

$$AR = 0.74$$

5. Design Example (For Canard Planform)

From previous analysis and development, the following parameters have been defined:

$$M = 3.0$$

$$S_c = 1.09 \text{ ft}^2$$

$$(t/c)_c = 0.03$$

If the leading edge sweep is set 5 degrees behind the sonic value and the taper ratio is set to zero, the following planform table results.

$$\Delta_{LE} = 75.5^\circ$$

$$\lambda = 0$$

AR	b	c _r	c	K _{W(B)}	K _{B(W)}
2.5	1.63	1.34	.89	1.29	.52
2.0	1.48	1.47	.98	1.31	.55
1.5	1.28	1.70	1.14	1.35	.59
1.0	1.04	2.10	1.40	1.47	.76
.5	.74	2.94	1.96	1.51	.90
.25	.52	4.19	2.79	1.59	1.02

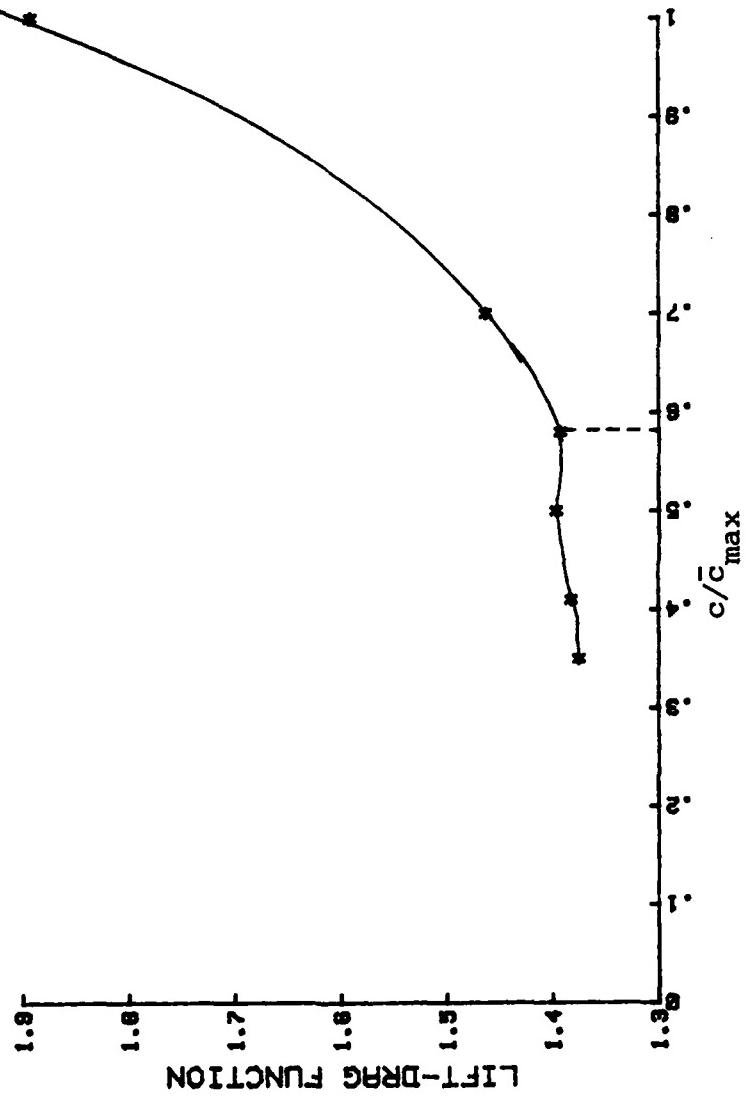


Figure (7-8). Wing lift-drag function.

AR	C_{D_f}	C_{D_0}	$AR \tan \Delta_{LE}$	$\frac{\beta}{\tan \Delta_{LE}}$	$C_{Na}(k)$
2.5	.0069	.0087	9.67	.73	
2.0	.0067	.0085	7.73	.73	2.7900
1.5	.0067	.0085	5.80	.73	3.6339
1.0	.0064	.0082	3.87	.73	2.5952
.5	.0061	.0079	1.93	.73	2.0880
.25	.0058	.0076	.97	.73	1.0124

AR	\bar{c}/\bar{c}_{max}	F_1	F_2	F
2.0	.35	1.0000	.3628	1.3628
1.5	.41	1.000	.3844	1.3844
1.0	.50	.9647	.3901	1.3557
.5	.70	.9294	.4849	1.4143
.25	1.00	.8941	1.0000	1.8941

From Figure (7-9) it is seen that there is a minimum of the lift-drag function at $c/\bar{c}_{max} = 0.5$. The optimum canard is then,

$$\bar{c} = 0.5 \bar{c}_{max} = 1.40 \text{ ft} = 16.80 \text{ in}$$

$$c_r = 2.09 \text{ ft} = 25.08 \text{ in}$$

$$b = 1.04 \text{ ft} = 12.48 \text{ in}$$

$$AR = 0.99$$

$$c = 2.01 \text{ ft} = 24.12 \text{ in}$$

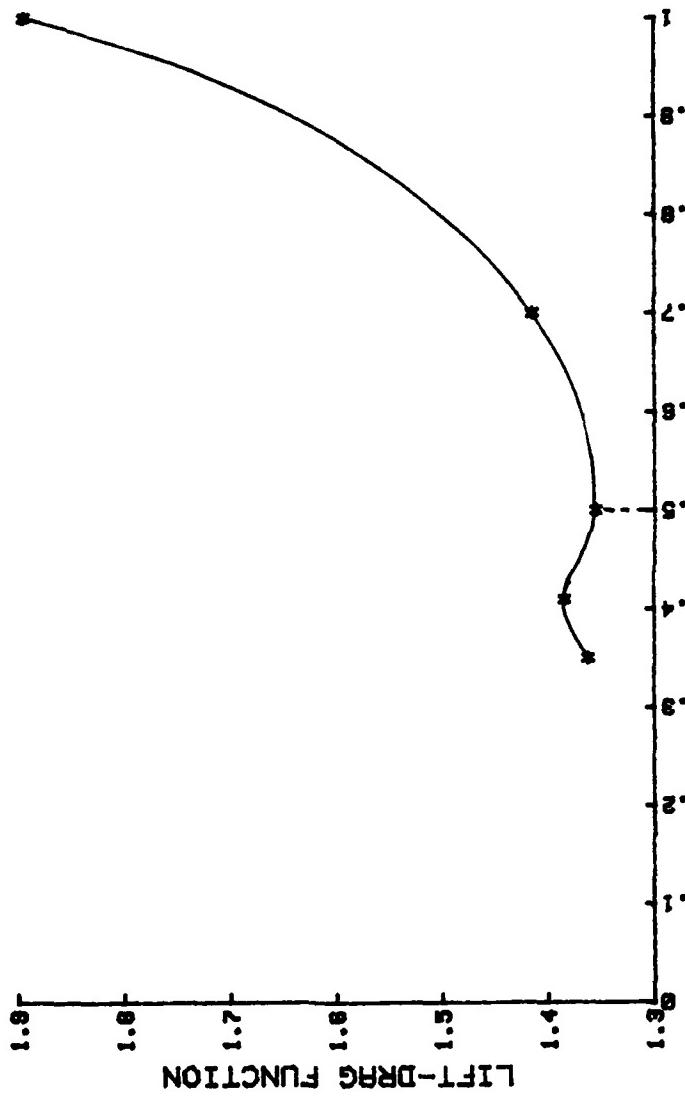


Figure (7-9). Canard lift-drag function.

E. DESIGN EXAMPLE (REVISION OF ZERO LIFT-DRAG AND LIFT-CURVE SLOPE)

Wing: $c = 3.62 \text{ ft}$

$$c_f = .0014$$

$$(C_{D_0})_w = .0750$$

Canard: $\bar{c} = 1.40 \text{ ft}$

$$c_f = .0016$$

$$(C_{D_0})_c = .0166$$

Body: $(C_{D_0})_B = .2292$

The complete missile C_{D_0} including the interference factor of 1.25, is now 0.4010. From the previous calculation of the baseline zero lift-drag coefficient, the drag has been reduced by 14.6 percent.

From equation (6) in Chapter 6 the lift-curve slope is now

$$C_{N\alpha} = 25.17/\text{rad}$$

Figure (7-10) is the missile design to this point.

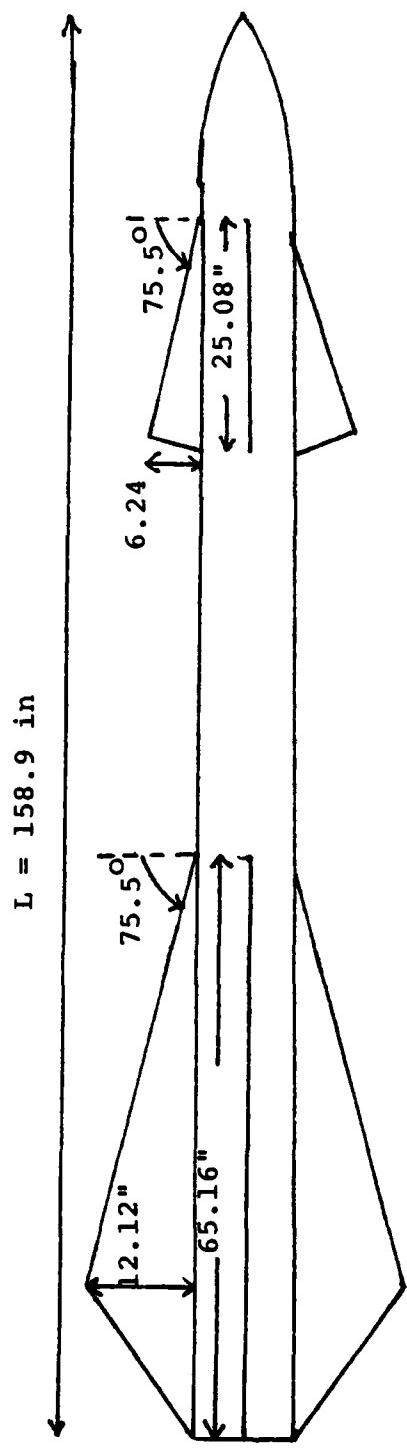


Figure (7-10). Design with "optimum" wings.

VIII. PROPELLION REQUIREMENTS

The following discussion presents a method for preliminary sizing of a solid rocket motor for a boost-sustain trajectory of an air launched or a surface launched tactical missile. The analysis consists of sizing the booster from incremental velocity considerations and sizing, the sustainer for the maximum range required at the operational altitude. The method assumes a constant acceleration boost and a constant altitude cruise.

A. BOOSTER INITIAL SIZING

Since the control system cannot respond properly while in the boost phase of flight, it becomes important to make the boost time as short as possible. The limiting factor for boost time is the maximum axial acceleration the airframe and components can withstand. This maximum acceleration is normally around 30 g's. The boost time then becomes,

$$t_b = \int_{v_1}^{v_2} \frac{dv}{a}$$

For constant acceleration,

$$t_b = \frac{v_2 - v_1}{a} = \frac{\Delta v_B}{a} \quad (1)$$

The incremental velocity during boost, Δv_B , can be derived from the equations of motion.

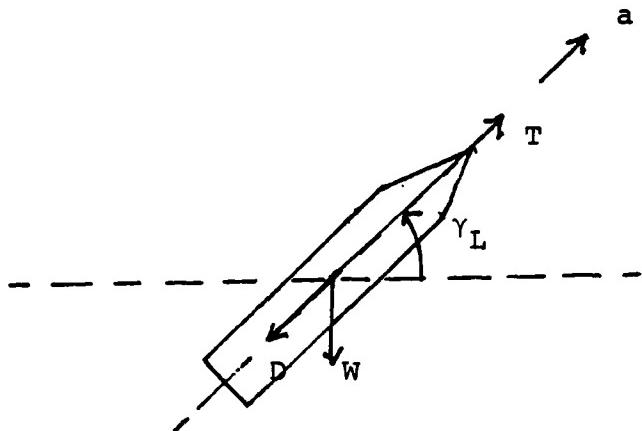


Figure (8-1). Forces acting on the missile.

If during the boost phase the thrust, drag and launch angle are considered constant, the axial acceleration of the missile is constant and may be written as:

$$m \frac{dv}{dt} = T - D - W \sin \gamma_L$$

$$dv = g \left(\frac{T-D}{W} \right) dt - g \sin \gamma_L dt$$

If the launch velocity is v_1 and the velocity at the end of boost is v_2 ,

$$\int_{v_1}^{v_2} dv = \int_0^{t_b} g \left(\frac{T-D}{W} \right) dt - \int_0^{t_b} g \sin \gamma_L dt \quad (2)$$

The vehicle weight in equation (2) is a function of time. If the propellant weight is given by W_p , and the propellant grain burn is linear with time, the missile weight can be expressed as,

$$W = W_L - W_p \left(\frac{t}{t_b} \right)$$

where W_L = Launch weight.

Equation (2) then becomes,

$$\Delta V_B = V_2 - V_1 = g(T - D) \int_0^{t_b} \frac{t_b dt}{(W_L t_b - W_p t)} - g \sin \gamma_L t_b$$

Integrating the first term yields,

$$\Delta V_B = \frac{g(T - D)t_b}{W_p} \ln \left[\frac{(W_L - W_p)t_b}{W_L t_b} \right] - g \sin \gamma_L t_b$$

Since the empty weight is given by

$$W_E = W_L - W_p$$

The incremental velocity can be written as

$$\Delta V_B = \frac{g(T - D)t_b}{W_p} \ln \frac{W_L}{W_E} - g \sin \gamma_L t_b \quad (3)$$

A first estimate of propellant weight can be made if the drag is assumed zero in equation (3), and $I_{sp} = Tt_b/W_p$

$$\Delta V_B = g I_{sp} \ln \frac{W_L}{W_E} - g \sin \gamma_L t_b \quad (4)$$

In the above expression the specific impulse, I_{sp} , is characteristic of the propellant chosen and can be found from historical data, or a specific propellant value may be used.

Equation (4) can then be solved for $\frac{W_L}{W_E}$ and the first estimate of propellant weight, W_p , can be found as follows,

$$W_p = W_L \left(1 - \frac{W_E}{W_L}\right)$$

With the propellant weight and specific impulse known, the total impulse and average thrust required for boost can be found.

$$I_T = I_{sp} W_p$$

$$T = \frac{I_T}{t_b}$$

With the thrust known and an average value of drag assumed, equation (3) can be iterated for an improved estimate of the propellant weight needed.

The final result is used to calculate the booster combustor volume and length.

$$V_B = W_p / \rho_p n_p$$

where n_p = volumetric packing factor

$$L_B = 4V_B / \pi D_C^2$$

The combustor diameter is limited by the missile maximum diameter and is a design choice. The propellant density, ρ_p , and volumetric packing factor, n_p , are determined from historical data or given for a specific propellant.

B. SUSTAINER INITIAL SIZING

The sustainer thrust required to maintain cruise is the driving factor in the sustainer motor sizing. For initial sizing purposes, a level, constant velocity cruise segment is assumed. In this analysis the thrust required is equal to the aerodynamic drag developed by the missile. The performance requirements determine the maximum range, operating altitude, and velocity of the missile. From these requirements the sustainer burn time, t_s , can be determined.

$$t_s = \frac{R_{\max} - S_B}{V_M}$$

where S_B = distance traveled during boost.

Since sustainer thrust is equal to drag, $T_s = D$, the total impulse required becomes,

$$(I_T)_S = D t_s$$

The sustainer nozzle is sized for the operational altitude of the missile. The thrust coefficient, C_{Fd} , can be expressed as;

$$C_{Fd} = C_d^\lambda \sqrt{\frac{2\gamma^2}{\gamma-1} \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - \left(\frac{p_0}{p_c}\right)^{\frac{\gamma-1}{\gamma}} \right]}$$

The constant, C_d , is the nozzle efficiency and has been determined from historical data to be 0.96. The nozzle half angle correction, λ , is given by;

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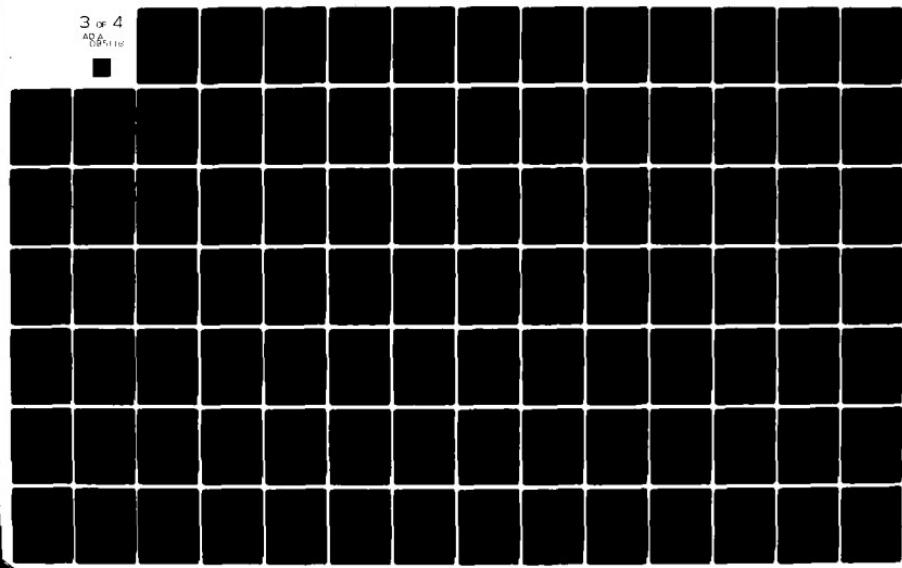
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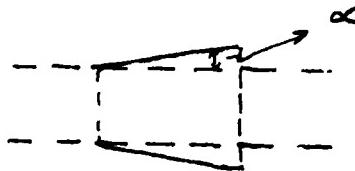
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$$\lambda = (1 + \cos \alpha)/2$$



where α is the nozzle half angle. A nominal value used in many designs is 15° . Larger values of α give significant non-axial flow components in the nozzle. Values less than 15° give large nozzle lengths and therefore contribute to excessive missile weights. The combustion chamber pressure is also a design choice. A lower usable value of P_c is 200 psia for sustainers and 500 psi for boosters. The chamber pressure cannot exceed the maximum expected operating pressure of the missile (MEOP).

The thrust of the sustainer can be expressed as a function of the thrust coefficient, chamber pressure, P_c , and throat area, A_t . The throat area required to deliver the required sustainer thrust can be determined from

$$T_s = C_{Fd} P_c A_t$$

The nozzle area ratio is a function of the pressure ratio P_0/P_c , where P_0 is the ambient pressure if the nozzle expands the flow fully.

$$\frac{A_e}{A_t} = \frac{1}{M_e} \left\{ \frac{2}{\gamma+1} \left[1 + \left(\frac{\gamma+1}{2} \right) M_e^2 \right] \right\}^{\frac{\gamma+1}{2(\gamma-1)}}$$

where

$$\frac{P_0}{P_c} = \left[1 + \left(\frac{\gamma-1}{2} \right) M_e^2 \right]^{-\frac{\gamma}{\gamma-1}}$$

From the previous equation the exit area can be determined.

From the throat and exit diameters, the nozzle length is found by

$$L_n = \frac{D_e - D_t}{2 \tan \alpha}$$

At this point in the design the nozzle exit area and length should be checked to see if they exceed any specifications on the missile; i.e., the nozzle exit area must be less than the base area of the missile.

Finally, the delivered specific impulse is typical of the propellant chosen for the sustainer. The propellant weight is then,

$$w_p = \frac{(I_T)_s}{I_{sp}}$$

The sustainer combustor volume and length are then determined in the same manner as the booster. The equations concerning the sustainer nozzle also apply to the booster nozzle with the appropriate booster areas, pressures and thrust substituted.

C. ROCKET MOTOR CHAMBER PRESSURE

As indicated earlier the delivered thrust increases with increasing chamber pressure, P_c ; however, the wall thickness in the rocket motor depends upon the expected operating pressure within the chamber. If the wall thickness is t inches, and the yield stress of the casing material is σ_y , it can be shown that the thickness required of the motor casing is given by,

$$t = \frac{P_{\max} r}{\sigma_y} \quad (5)$$

where r is the chamber radius, and P_{\max} is the maximum chamber pressure, which is taken to be $1.2 P_c$ to allow for variations within the propellant. From equation (5) the weight of the motor casing can be determined if the casing is cylindrical and $t \ll r$

$$W_c = \rho_c V_c = 2\pi r l_c t \rho_c$$

Substituting for t

$$W_c = \frac{2.4\pi r^2 l_c \rho_c P_c}{\sigma_y} \quad (6)$$

The specific impulse of the rocket motor is given by.

$$I_{sp} = \frac{V_i}{g_c} = \frac{1}{g_c} \left\{ 2g_c \left(\frac{\gamma}{\gamma-1} \right) R T_0 \left[1 - \left(\frac{P_0}{P_c} \right)^{\frac{\gamma-1}{\gamma}} \right] \right\}^{1/2} \quad (7)$$

From equations (6) and (7), it can be seen that a compromise must be made in selecting the chamber pressure; since increasing P_c increases the specific impulse but also increases the casing weight. An attempt must be made to find an optimum operating pressure.

If the rocket motor weight consists of the nozzle, propellant and casing,

$$W_M = W_N + W_P + W_C \quad (8)$$

and the total impulse, which is constant is given by,

$$I = Tt_b = I_{sp} W_p$$

The optimum chamber pressure can be found by minimizing the weight and maximizing the impulse.

$$\frac{d}{dp_c} \left(\frac{W_M}{I} \right) = \frac{d}{dp_c} \left(\frac{W_M}{I_{sp} W_p} \right) = 0$$

differentiating yields,

$$\frac{1}{I_{sp}} \frac{d}{dp_c} \left(\frac{W_M}{W_p} \right) - \frac{W_M}{W_p} \frac{d(1/I_{sp})}{dp_c} = 0 \quad (9)$$

$$\frac{1}{I_{sp}} \frac{d}{dp_c} \left(\frac{W_M}{W_p} \right) - \frac{W_M}{W_p^2} \frac{d I_{sp}}{dp_c} = 0$$

From equation (8)

$$\frac{W_M}{W_p} = \frac{W_c}{W_p} + \frac{W_N}{W_p} + 1 \quad (10)$$

Substituting equation (10) into equation (9),

$$\frac{1}{W_p} \frac{dW_c}{dp_c} - \frac{W_c}{W_p^2} \frac{dW_p}{dp_c} - \frac{W_N}{W_p^2} \frac{dW_p}{dp_c} - \frac{W_M}{W_p I_{sp}} \frac{d I_{sp}}{dp_c} = 0$$

Substituting equations (12) and (13) into equation (11)

$$\frac{2.4 \rho_c w_p}{\rho_p \sigma_y} - \frac{w_p R T_0}{g_c I_{sp}^2 p_c} \left(\frac{p_0}{p_c} \right)^{\frac{\gamma-1}{\gamma}}$$

Substituting for $g_c I_{sp}^2$ in the above equation yields,

$$\frac{\gamma-1}{2\gamma} \frac{1}{\left[\left(\frac{p_0}{p_c} \right)^{(1-\gamma)/\gamma} - 1 \right] p_c} - \frac{2.4}{\rho_p} \left(\frac{\rho_c}{\sigma_y} \right) = 0 \quad (14)$$

If a propellant and casing material are chosen, equation (14) can be solved for the chamber pressures, p_c , which minimizes w_M/I . One interesting feature of equation (14) is that there is a minimum yield strength to density ration $\frac{\sigma_y}{\rho_c}$ that will yield a usable chamber pressure, furthermore this optimization yields a very shallow minimum. Therefore, if the thrust or exit diameter requires an increase in p_c the penalty paid in additional casing weight will be small.

D. TYPICAL PROPELLANTS

From the preceding discussion it can be seen that some knowledge of propellants to be used in the rocket motor is needed. The thrust developed by a rocket motor depends directly upon the pressure in the chamber.

$$T = C_F A_T p_c$$

Since

$$w_p = \frac{I}{I_{sp}} = \frac{Tt_b}{I_{sp}}$$

$$\frac{dw_c}{dp_c} + (w_c + w_N) \frac{I_{sp}}{2} \frac{dI_{sp}}{dp_c} - \frac{w_M}{I_{sp}} \frac{dI_{sp}}{dp_c} = 0$$

From equation (8)

$$w_c + w_N = w_M - w_p$$

Substituting for $w_c + w_N$

$$\frac{dw_c}{dp_c} - \frac{w_p}{I_{sp}} \frac{dI_{sp}}{dp_c} = 0$$

From equation (6)

$$\frac{dw_c}{dp_c} = \frac{2.4\pi r^2 l_c \rho_c}{\sigma_y} \quad (11)$$

Since

$$w_p = \pi r^2 l_c \rho_p$$

$$\frac{dw_c}{dp_c} = \frac{2.4\pi \rho_c w_p}{\rho_p \sigma_y} \quad (12)$$

from equation (3)

$$\frac{dI_{sp}}{dp_c} = \frac{RT}{g_c p_c I_{sp}} \left(\frac{p_o}{p_c} \right)^{\frac{y-1}{y}} \quad (13)$$

Because of this relationship, high pressures are needed for the boost phase, and relatively low pressure are needed for the sustain phase of flight. The pressure developed in a rocket motor chamber is a function of the burn rate and burn surface area of the propellant. In order to provide more burn area for the boost phase, the grain normally has an internal star perforation; while a sustain motor is normally an end burning grain of solid cross-section. The volumetric loading of a rocket motor is defined as the ratio of the propellant volume to the rocket motor chamber volume.

$$n_p = \frac{\text{Grain volume}}{\text{Chamber volume}}$$

Due to erosive burning effects the volumetric loading of a booster is normally limited to less than 0.9. For efficient packing the ratio varies from 0.7 to 0.9. The volumetric loading of an end burning sustainer engine is normally 1.0. The range of propellant characteristics is shown in Table (8-I).

TABLE (8-I). Typical Propellant Properties [14,15].

	<u>Sustainer</u>	<u>Booster</u>
I_{sp} (sec)	180-210	210-260
ρ_p (lbm/in ³)	.059-.062	.062-.065
γ	1.24 - 1.27	1.22 - 1.26
n_p	1.00	0.85

E. DESIGN EXAMPLE (BOOSTER)

From the previous analysis the launch speed is $M = 1.5$, and the cruise speed is $M = 3.0$ at an altitude of 10,000 feet. The boost incremental velocity is then,

$$\Delta V_B = 1616.10 \text{ ft/sec}$$

If the maximum acceleration during boost is a constant 30 g's (assume sea level g_C), the time of burn is,

$$t_b = \Delta V_B / a_{\max} = \frac{1616.10 \text{ ft/sec}}{30(32.2 \text{ ft/sec}^2)}$$

$$t_b = 1.67 \text{ sec}$$

From equation (4) with $\gamma_L = 0$

$$\ell_n \frac{W_L}{W_E} = \frac{\Delta V_B}{g I_{sp}}$$

Assume $I_{sp} = 250 \text{ sec}$ from Table (I)

$$\ell_n \frac{W_L}{W_E} = 0.2008$$

$$W_p = W_L (1 - \frac{W_E}{W_L})$$

For $W_L = 794.83$

$$W_p = 144.57 \text{ lbm}$$

The thrust provided is

$$(T)_b = \frac{I_{sp} W_p}{t_b} = 21,642 \text{ lb}_f$$

Since the missile flies essentially at zero angle of attack during boost, the drag at the end of boost is,

$$D = (C_{D_0}) q S_{ref}$$

$$D = 2006 \text{ lb}_f$$

Therefore if an average drag of 1000 lbs is assumed, equation (3) can be used to find a new W_L/W_E

$$\ell_n \frac{W_L}{W_E} = \frac{\Delta V_B W_p}{g(T - D)t_b}$$

$$\frac{W_L}{W_E} = .2270$$

$$W_p = 161.40 \text{ lb}_m$$

This gives a thrust of 24,162 lbs. One more iteration of equation (4) gives,

$$W_p = 159.46 \text{ lb}_m$$

$$(T)_s = 23,872 \text{ lb}_f$$

Continued iteration of equation (4) does not change the propellant weight.

From Table (I); $\rho_p = .062 \text{ lb}_m/\text{in}^3$

$$n_p = .85$$

The booster volume then becomes

$$V_B = 3025.81 \text{ in}^3$$

Allowing one-half inch for structure, the booster length becomes.

$$L_B = \frac{\frac{4}{\pi} \frac{V_B}{D_C^2}}{2} = 42.69 \text{ inches}$$

F. DESIGN EXAMPLE (SUSTAINER)

The maximum range of the missile was determined to be 35 nmiles.

$$R_{\max} = 212,800 \text{ ft}$$

The distance traveled during boost is

$$S_B = \bar{v}_M t_b$$

$$\bar{v} = \frac{v_1 + v_2}{2} = 2424.15 \text{ ft/sec}$$

$$S_B = 4048.33 \text{ ft}$$

Therefore, at a cruise Mach number of 3.0, the sustainer time of burn is

$$t_s = 64.59 \text{ sec.}$$

The cruise drag consists of zero-lift drag and induced drag.
The lift coefficient of the missile minus the booster is;

$$C_L = \frac{W}{q S_{ref}}$$

$$C_L = 0.1270$$

From Chapter 6

$$C_D = C_{D_0} + C_L^2 / C_{Na}$$

$$C_D = .4016$$

The cruise drag then becomes

$$D = 2003 \text{ lb}_f$$

The total impulse required is then;

$$(I_T)_s = D t_s = 129,761 \text{ lb}_f \cdot \text{sec}$$

From Table (I) the specific impulse is 210 sec; therefore,
the weight of propellant needed for cruise is

$$(w_p)_s = \frac{(I_T)_s}{I_{sp}} = 618 \text{ lb}_m$$

The volume required is

$$V_s = 9967.7 \text{ in}^3$$

The sustainer length becomes,

$$L_s = \frac{4V_s}{\pi D_c^2} = 140.62 \text{ inches}$$

The total rocket motor length is, L_{RM} ;

$$L_{RM} = L_B + L_S$$

$$L_{RM} = 183.31 \text{ in}$$

The rocket motor alone exceeds the total length of the baseline missile. This will lead to large length to diameter ratios for the entire missile; therefore, the rocket motor length must be decreased. That is the subject of the next section.

G. REDUCING ROCKET MOTOR LENGTH

If the rocket motor length of the last section leads to excessive length to diameter ratios for the missile, the motor length must be decreased. This may be accomplished in one of two ways: (1) If the mission profile has a long cruise segment, the cruise altitude may be increased thereby decreasing the drag and the total impulse required. (2) If the cruise altitude cannot be varied, the missile diameter must be resized for propulsion considerations. In this design example option (2) will be selected.

The missile length can be expressed as

$$L = L_n + L_G + L_C + L_{WH} + L_{RM} \quad (15)$$

L_n = Nose length
 L_G = Guidance section length
 L_c = Control section length
 L_{WH} = Warhead section length
 L_{RM} = Rocket motor length

The nose and warhead sections have been previously defined. Since the propellant weight for the range requirement is known, the diameter necessary can be determined if a maximum missile length is specified. The drag will increase slightly due to the increased diameter but the total impulse, and therefore the propellant weight, will change only slightly.

The rocket motor length can be expressed in terms of the diameter by

$$L_{RM} = \frac{4W_p}{\rho_p \pi D^2}$$

Equation (15) then becomes

$$(L) = L_n + L_G + L_c + L_{WH} + \frac{4W_p c}{\rho_p \pi D^2}$$

The guidance and the control sections are each normally about 10 percent of the missile length.

$$(L) = \left(\frac{L_n}{D}\right)D + .1L + .1L + \left(\frac{L_{WH}}{D}\right)D + \frac{4W_p}{\rho_p \pi D^2}$$

$$\left[\frac{L_n}{D} + \frac{L_{WH}}{D}\right] D^3 - .8LD^2 + \frac{4W_p}{\rho_p \pi} = 0 \quad (16)$$

As mentioned in Chapter 2 the missile is normally designed for an existing launcher. This launcher will have a maximum length that it can accomodate, L_{\max} . If this value is substituted into equation (16), the required diameter can be found.

1. Design Example (Resizing for Propulsion)

From previous analysis; $L_n/D = 2.23$

$$L_{WH}/D = 2.50$$

The propellant weight from before was, $w_p = 777.46$ lbs. If the maximum length of the missile is taken as 210 inches, equation (16) becomes,

$$4.73D^3 - 168D^2 + 15,467.08 = 0.0$$

The required diameter is then,

$$(D)_{req} = 11.75 \text{ inches}$$

Allowing 0.25 in for structure, the missile diameter becomes 12 inches. From the equations for the rocket motor lengths,

$$L_b = 26.62 \text{ in}$$

$$L_s = 90.47 \text{ in}$$

$$L_{RM} = 117.09 \text{ in}$$

H. DESIGN EXAMPLE (CHAMBER PRESSURE)

From equation (14), the strength to density ratio can be solved for in terms of p_c .

$$\left(\frac{\sigma_y}{\rho_c}\right)_{\min} = \frac{2\gamma}{(\gamma-1)} \left[\left(\frac{p_0}{\rho_p} \right)^{\frac{1-\gamma}{\gamma}} - 1 \right] p_c \left(\frac{2 \cdot 4}{\rho_p} \right)$$

If the minimum usable chamber pressure is $p_c = 500$ psi, and using the propellant previously selected.

$$\gamma = 1.24$$

$$\rho_p = .062$$

$$p_0 = 10.11$$

the minimum strength to density required can be found.

$$\left(\frac{\sigma_y}{\rho_c}\right)_{\min} = 225,542.87 \text{ in}$$

Inconel 718 is a common metal used in combustion chambers and has the following properties [4]:

$$\sigma_y = 180,000 \text{ psi}$$

$$\rho_c = .2662 \text{ lb/in}^3$$

$$\frac{\sigma_y}{\rho_c} = 676,183 > \left(\frac{\sigma_y}{\rho_c}\right)_{\min}$$

The optimum chamber pressure is then from equation (14)

$$p_c = 1132.55 \text{ psi}$$

The rocket motor chamber wall thickness is given by equation (5) as

$$t = \frac{p_{\max} r}{\sigma_y}$$

$$t = 0.05 \text{ inches}$$

The wall weight is given by equation (2)

$$W_c = 2\pi r l_{RM} t \rho_c$$

$$W_c = 59.5 \text{ lb}_m$$

I. DESIGN EXAMPLE (SIZING THE NOZZLE)

For initial analysis it is assumed that the flow is fully expanded, in a nozzle with a half angle of 15 degrees. With the chamber pressure of 1132.55 psi the thrust coefficient at altitude is

$$C_{F_d} = C_d \lambda \sqrt{\frac{2\gamma^2}{\gamma-1} \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}} \left[1 - \left(\frac{p_0}{p_c}\right)^{\frac{\gamma-1}{\gamma}}\right]}$$

$$C_{F_d} = 2.3023$$

The throat area and diameter then become

$$A_t = \frac{T}{C_{F_d}} p_c = 9.1552 \text{ in}^2$$

$$d_t = 3.4142 \text{ in}$$

From isentropic tables for $p_0/p_c = .0089$

$$\frac{d_t}{d_e} = \left(\frac{A_t}{A_e}\right)^{1/2} = .3342$$

The exit diameter becomes,

$$d_e = 10.22 \text{ in}$$

The nozzle length for a 15 degree half angle is,

$$l_n = 12.70 \text{ in}$$

The nozzle is shown in Figure (8-2). Figure (8-3) is the design with resized diameter and length for propulsion and the optimum wings determined in Chapter 7.

J. TAIL SIZING

As mentioned in Chapter 7 the tail (canard) sizing is accomplished for zero static stability at launch. Now that the missile length has been fixed, the canard can be sized. Figure (8-3) is a drawing of the design to this point. The total lifting surface required is

$$S_W + S_C = 6.57 \text{ ft}^2$$

if $k = S_C/S_W$ (16)

From equation (5) in Chapter 7, for $C_{M\alpha} = 0$

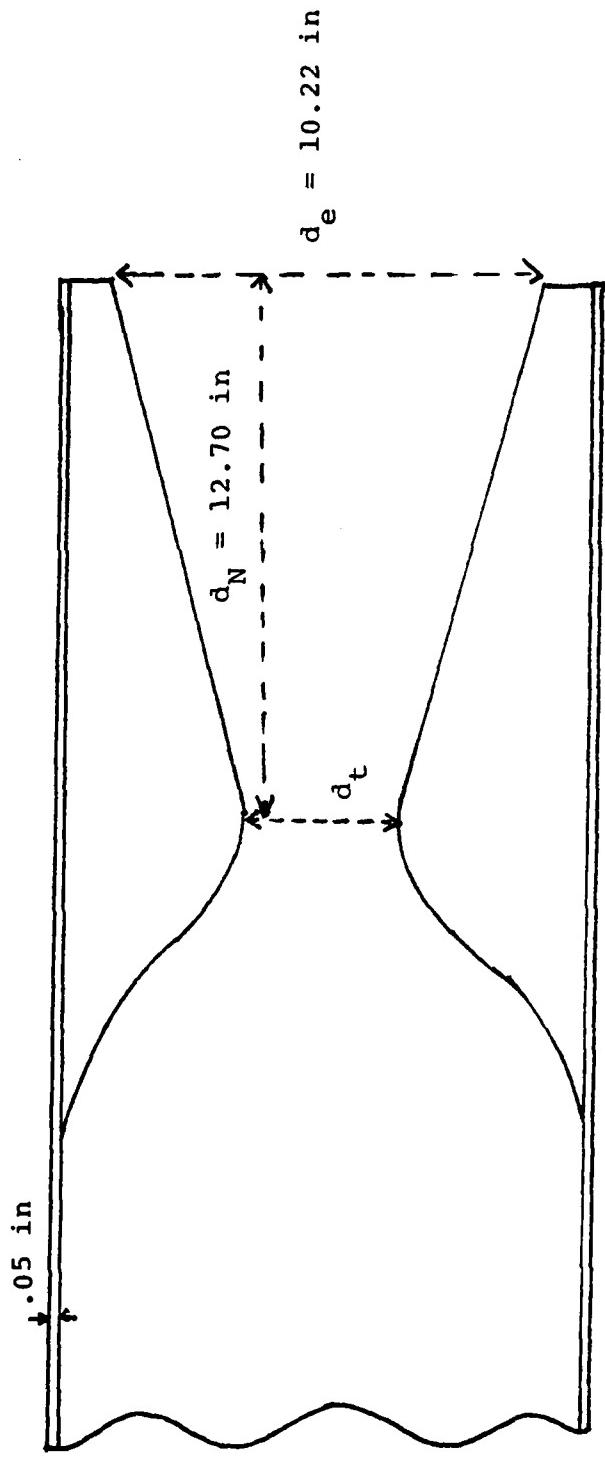


Figure (8-2). Nozzle geometry.

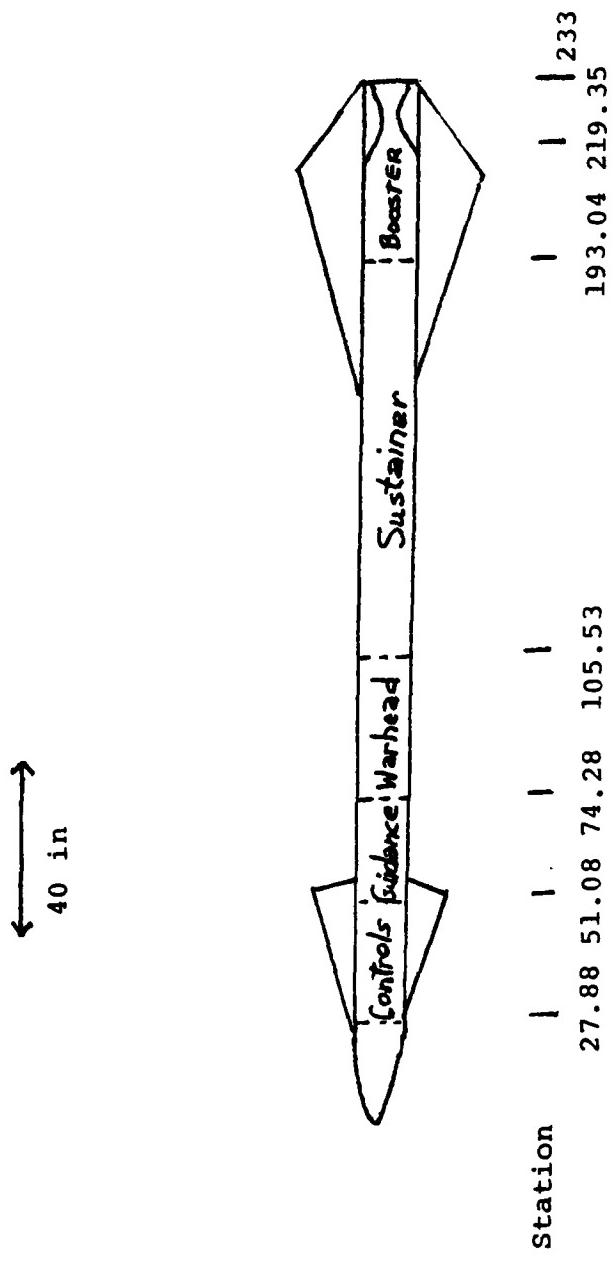


Figure (8-3). Missile design.

$$0 = \left(C_{Na} \right)_N + \left(C_{Na} \right)_W x_W \frac{S_W}{S_{ref}} (K_{W(B)} + K_{B(W)}) \\ (17)$$

$$+ \left(C_{Na} \right)_C x_C \frac{S_W}{S_{ref}} (K_{C(B)} + K_{B(C)})$$

Equations (16) and (17) above can be solved for S_W and k .

$$S_W = 4.70 \text{ ft}^2$$

$$k = 0.40$$

Therefore, $S_C = 1.87$

If the same aspect ratio as found in Chapter 7 is used, the wing and canard planforms become,

Wing: $b_w = 1.86 \text{ ft}$	Canard: $b_c = 1.36 \text{ ft}$
$(c_r)_w = 5.05 \text{ ft}$	$(c_r)_c = 2.75 \text{ ft}$
$(c)_w = 3.60 \text{ ft}$	$(c)_c = 2.63 \text{ ft}$

IX. NONLINEAR AERODYNAMICS AND AERO1

As mentioned in Chapter 5 the analysis thus far has been for one speed and small angles of attack. The aerodynamics change dramatically from subsonic to supersonic flow and with increasing angle of attack. This chapter presents the methods used primarily by USAF Stability and Control DATCOM and covers all configurations and flight speeds. As will be shown, the method becomes very involved and therefore lends itself to programming on a digital computer.

The lifting characteristics of both wings and bodies become nonlinear as the missile angle of attack increases above 10° . Up to this point in the thesis development only the linear contribution has been considered. For large angles of attack, the nonlinear term, which is due to flow separation, must be considered. The relative effect of these terms on the normal force and pitching moment coefficients is shown in Figure (9-1).

A. VISCOUS CROSS-FLOW

As can be seen in Figure (9-1), at large angles of attack the forces acting are primarily nonlinear. The nonlinear term is normally described through the use of the viscous cross-flow coefficient, C_{d_c} .

$$C_{d_c} = \frac{f_v}{q_n s_p} \quad s_p = \text{planform area.}$$

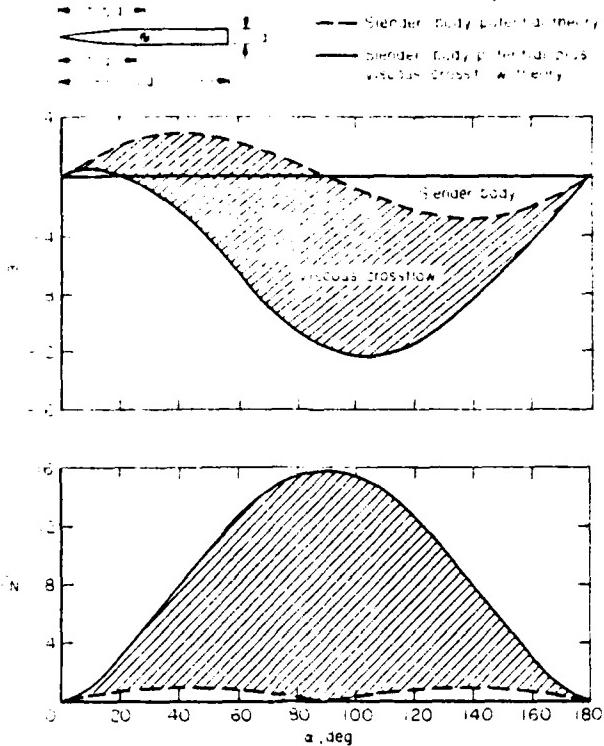


Figure (9-1). Relative contributions of linear and nonlinear terms [19].

For an infinitely long circular cylinder, the viscous cross-flow force per unit length is

$$f_v = 2r C_{d_c} q_n \quad (1)$$

The viscous cross-flow coefficient is determined experimentally and is a function of normal Mach number and Reynolds number. Figure (9-2) gives the cross-flow drag coefficient as a function of cross-flow Mach number.

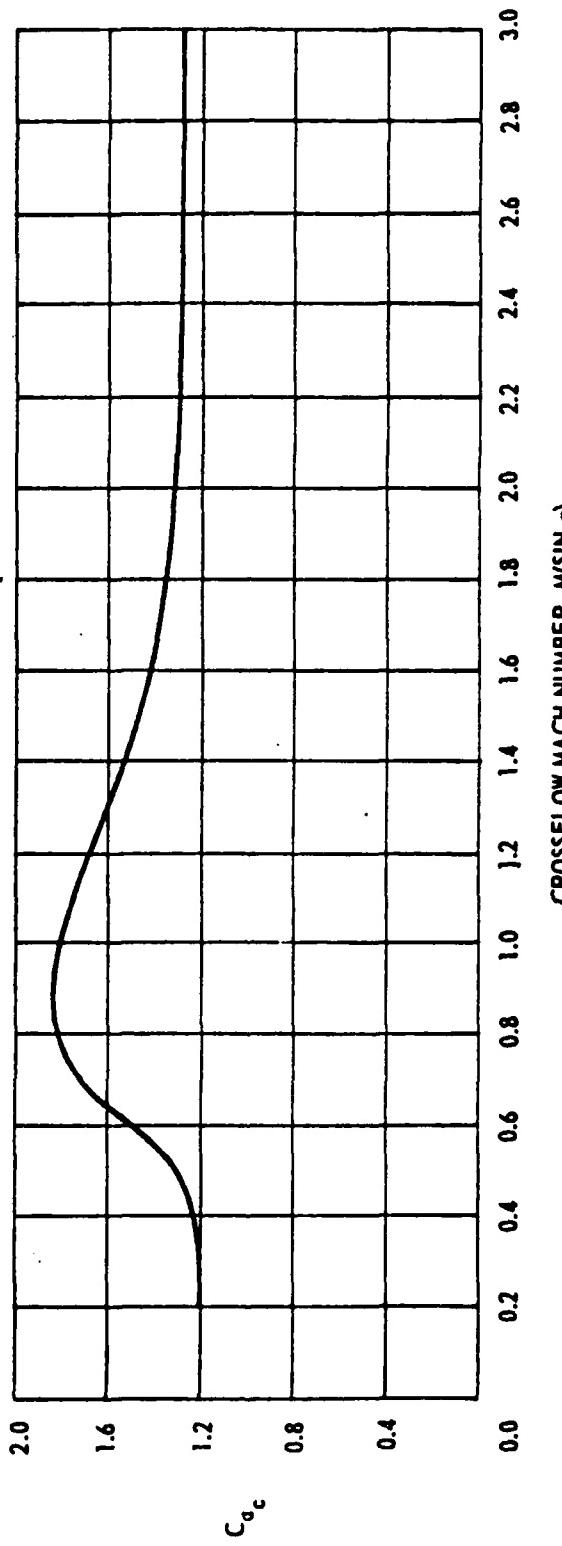


Figure (9-2). Cross-flow drag coefficient [18].

$$M_n = M_M \sin \alpha$$

$$R_{e_n} = R_e \sin \alpha$$

$$V_n = V_M \sin \alpha$$

Since in real flow there is spillage around the ends of a finite length cylinder, the drag is less than that for an infinite, 2-D cylinder. The cross-flow drag proportionality constant, n , is the ratio of the drag coefficient of a finite cylinder to the drag coefficient of an infinite cylinder.

$$n = \frac{C_{d_c}}{(C_{d_c})_{2-D}}$$

The proportionality constant is also determined experimentally and is given in Figure (9-3). Notice that the cross-flow drag proportionality constant approaches one as the length to diameter ratio becomes large, or the 2-D situation is approached.

The viscous cross-flow force per unit length for a finite cylinder then becomes

$$f_v = 2rnC_{d_c} \frac{\rho V_n^2}{2} \quad (2)$$

Since

$$\frac{\rho V_n^2}{2} = \frac{\rho V_M^2}{2} \sin^2 \alpha = q \sin^2 \alpha$$

equation (2) becomes,

$$f_v = 2nC_{d_c} q r \sin^2 \alpha$$

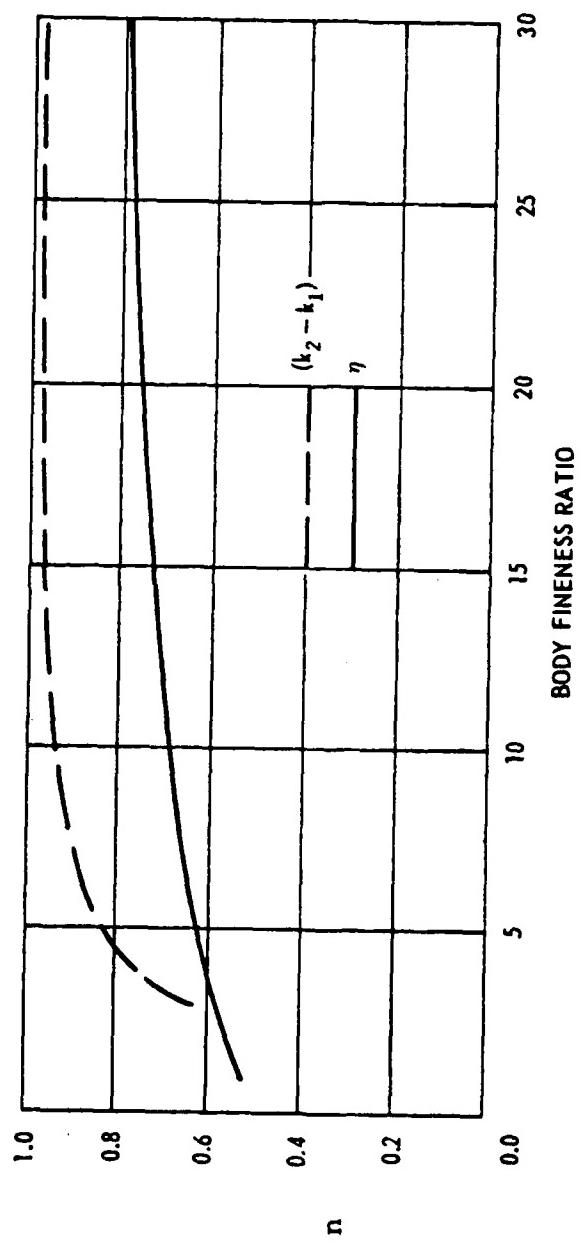


Figure (9-3). Cross-flow drag proportionality constant.

The viscous contribution to the normal force acting on the cylinder then becomes

$$(C_N)_V \equiv \frac{F_v}{q S_{ref}} = \frac{2n C_d \sin^2 \alpha}{S_{ref}} \int_0^l r dx$$

For a uniform cylinder,

$$(C_N)_V = n C_d \frac{S_p}{S_{ref}} \sin^2 \alpha$$

If this force is added to the term predicted by slender body potential theory, the total normal force acting on the body can be found. Reference [19] gives this total force as;

$$(C_N)_B = \frac{S_b}{S_{ref}} \sin^2 \alpha \cos \frac{\alpha}{2} + n C_d \frac{S_p}{S_{ref}} \sin^2 \alpha$$

Similar terms can be developed for the wing and tail since they also exhibit nonlinear behavior at high angles of attack. Care must be taken to separate the lift of Chapter 5 into the lift acting on the wing panel and the additional lift on the body before the cross-flow term is applied. The nonlinear cross-flow component of wing lift is caused by flow separation and the formation of vortices on the upper surface of the wing. This viscous term is given by, $C_{N_W(V)}$.

$$C_{N_W(V)} = C_{d_n} \sin^2 \alpha \frac{S_w}{S_{ref}}$$

Where C_{d_n} is the cross-flow drag coefficient for a flat plate and is given in Figure (9-4).

B. TOTAL MISSILE LIFT

The remainder of this chapter is a summary of references [17] and [18]. With the addition of the force due to viscous cross-flow, the total lift on the body becomes

$$C_{N_B} = (C_N)_B + C_{N_B(W)} + C_{N_B(T)-\alpha} + C_{N_B(T)-\delta}$$

where,

$(C_N)_B$ = linear and nonlinear lift force on the body

$C_{N_B(W)}$ = additional lift on the body due to the wing

$C_{N_B(T)-\alpha}$ = additional lift on the body due to the tail
(due to angle of attack)

$C_{N_B(T)-\delta}$ = additional lift on the body due to the tail
(due to control deflection)

The additional lift on the body in the presence of the wing and tail can be found as described in Chapter 6.

The wing lift is now composed of a linear and a nonlinear component and becomes

$$C_{N_W} = C_{N_W(B)} + C_{N_W(V)}$$

$$C_{N_W} = K_{W(B)} (C_{N_{ref}})_W \frac{S_W}{S_{ref}} \sin \alpha + C_{d_n} \sin^2 \alpha \frac{S_W}{S_{ref}}$$

The tail lift is found in a similar manner only with the additional term for any control surface deflection, δ .

$$C_{N_T} = C_{N_T(B)-\alpha} + C_{N_T(B)-\delta} + C_{N_T(V)}$$

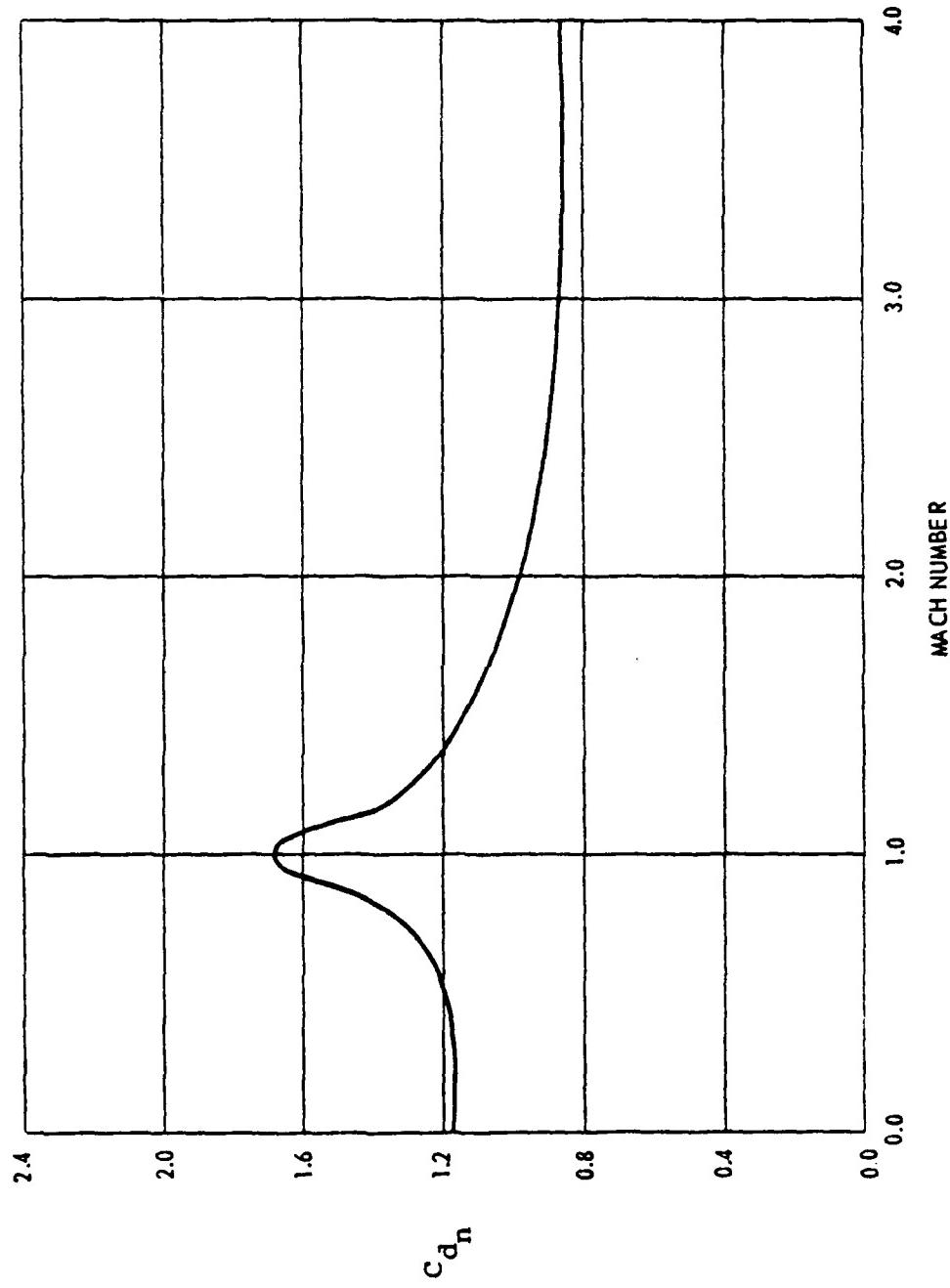


Figure (9-4). Drag coefficient for a flat plate normal to flow [18].

Notice that the linear terms in the above equations are multiplied by $\sin\alpha$ instead of α as in Chapter 6. The linear theory from which the Chapter 6 equations were derived assumes small angles of attack and therefore $\sin\alpha \approx \alpha$. This is not true at higher angles of attack and is therefore included here.

The total normal force then becomes, C_N

$$C_N = C_{N_B} + C_{N_W} + C_{N_T} + C_{L_i}$$

The last term, C_{L_i} , is the lift-loss due to downwash and is given by

$$C_{L_i} = \frac{(C_{Lx})_W (C_{L\alpha})_T [k_{W(B)} \sin\alpha + k_{W(B)} \sin\delta] i (b - r)_T s_w}{2\pi AR_T (f_w - r_w) s_{ref}}$$

where, i = interference factor

f_w = votex location

the above terms are found by the methods of reference [13].

C. DRAG CHARACTERISTICS OF A MISSILE

The total drag acting on a missile is the sum of the zero-lift drag and the induced drag (due to angle of attack and/or control surface deflection). The zero lift drag of bodies and wings is highly dependent on the missile speed.

Three speed regimes are normally considered.

Subsonic $\longrightarrow 0 \leq M_M < .8$

Transonic $\longrightarrow .8 \leq M_M \leq 1.2$

Supersonic $\longrightarrow M > 1.2$

Because of the empirical nature of the formulas for zero-lift drag, the procedures followed are essentially those of DATCOM [16].

1. Zero Lift Drag

a. Subsonic

At subsonic speeds the zero-lift drag consists of skin friction (incompressible) and pressure drag. The pressure drag at subsonic speeds is usually small compared to the drag due to skin friction. The entire zero-lift drag of a missile at subsonic speeds is given by;

$$C_{D_0} = C_{D_{0B}} + C_{D_{0W}} + C_{D_{0T}}$$

Where the components are the body, wing and tail contributions.

(1) Body drag, $C_{D_{0B}}$. The body zero-lift drag is given by;

$$C_{D_{0B}} = 1.02 C_{f_B} \left[1 + \frac{1.5}{f^{3/2}} + \frac{7}{f^3} \right] \frac{(S_{Wet})_B}{S_{ref}} + C_{A_B}$$

where C_{A_B} is found by the methods of Chapter 5.

The first term is the skin friction contribution and the next two terms are the pressure contributions. f is the body fineness ratio and is given by;

$$f = \frac{L}{D}$$

(2) Wing drag, $C_{D_{0W}}$. The wing zero-lift drag is given by,

$$C_{D_{0W}} = C_{f_w} \left[1 + 2\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4 \right] \frac{(S_{Wet})_w}{S_{ref}}$$

$(S_{Wet})_w$ is the entire wetted area of the wings. The thickness to chord ratio is given by (t/c) .

(3) Tail drag, $C_{D_{0T}}$. The tail zero-lift drag is given by,

$$C_{D_{0T}} = C_{f_T} \left[1 + 2\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4 \right] \frac{(S_{Wet})_T}{S_{ref}}$$

The above analysis assumes fully turbulent flow along all surfaces. C_{f_B} , C_{f_w} and C_{f_T} are the local average skin friction coefficients based on the local Reynolds number, R_e .

The reference lengths are the body length and wing/tail mean aerodynamic chords. The skin friction coefficient is then given by

$$C_f = \frac{4.55}{(\log_{10} R_e)^{2.58}}$$

b. Transonic

The total transonic zero-lift drag is composed of skin-friction drag, transonic wave drag, pressure drag and base pressure drag. The compressible skin friction drag is found by using a correction factor on the incompressible coefficient, C_{f_B} , found for subsonic flow. The compressibility

correction is found by the methods of Chapter 6. The skin friction drag then becomes

$$C_{D_{fB}} = 1.02 C_{f_c} \frac{(S_{Wet})_B}{S_{ref}}$$

The subsonic pressure drag is the same as before

$$C_{D_{pB}} = 1.02 C_{f_B} \left[\frac{1.5}{f^{3/2}} + \frac{7}{f^3} \right] \frac{(S_{Wet})_B}{S_{ref}}$$

The above equation applies for Mach numbers in the range of .8 to 1.0. The pressure drag then decreases linearly to zero at a Mach number of 1.2. The transonic wave drag $C_{D_{rB}}$ is determined using Figure (9-5), which is a function of nose fineness ratio $L_{N/D}$.

The total transonic zero-lift drag of the body then becomes

$$C_{D_{0B}} = C_{D_f} + C_{D_{pB}} + C_{D_{VB}} \frac{S_N}{S_{ref}} + C_{A_B}$$

where S_N is the cross sectional area at the nose-body junction.

The total transonic zero-lift drag of the aerodynamic surfaces is composed of the skin friction drag and a drag increment, ΔC_{D_0} , which is the transonic wave drag.

Experimental results show little increase in the viscous drag of aerodynamic surfaces from the subsonic to the transonic regimes; therefore the skin friction and pressure drag is the same as for subsonic flow and is given by

$$C_{D_{0W}} = C_{f_w} \left[1 + 2\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4 \right] \frac{S_{Wet}}{S_{ref}}$$

The wave drag of the aerodynamic surfaces if found from Figure (9-6) and is a function of $\frac{t}{c}$, AR and M. For swept wings the Mach number used in Figure (9-6) is given by,

$$M' = M [\cos \Delta c/4]^{1/2}$$

The transonic wave drag increment is then given by

$$\Delta C_{D_{0W}} = \Delta C_{D_{0W}'} [\cos \Delta c/4]^{2.5} \frac{S_w}{S_{ref}}$$

where $\Delta C_{D_{0W}'}$ is obtained from Figure (9-6). The tail zero-lift drag is found in the same manner as the wing. The total zero-lift drag (transonic) is then given by;

$$C_{D_0} = C_{D_{0B}} + C_{D_{0W}} + \Delta C_{D_{0W}} + C_{D_{0T}} + \Delta C_{D_{0T}}$$

c. Supersonic

The supersonic zero-lift drag of a missile is determined empirically by assuming a parabolic variation of C_{D_0} with Mach number between 1.2 and 3.0. The resulting equation is for the entire missile and is given by

$$C_{D_0} = C_{D_0'} - 1.7209 (C_{D_0''} - C_{D_0'}) + 1.5708 (C_{D_0''} - C_{D_0'}) \sqrt{M}$$

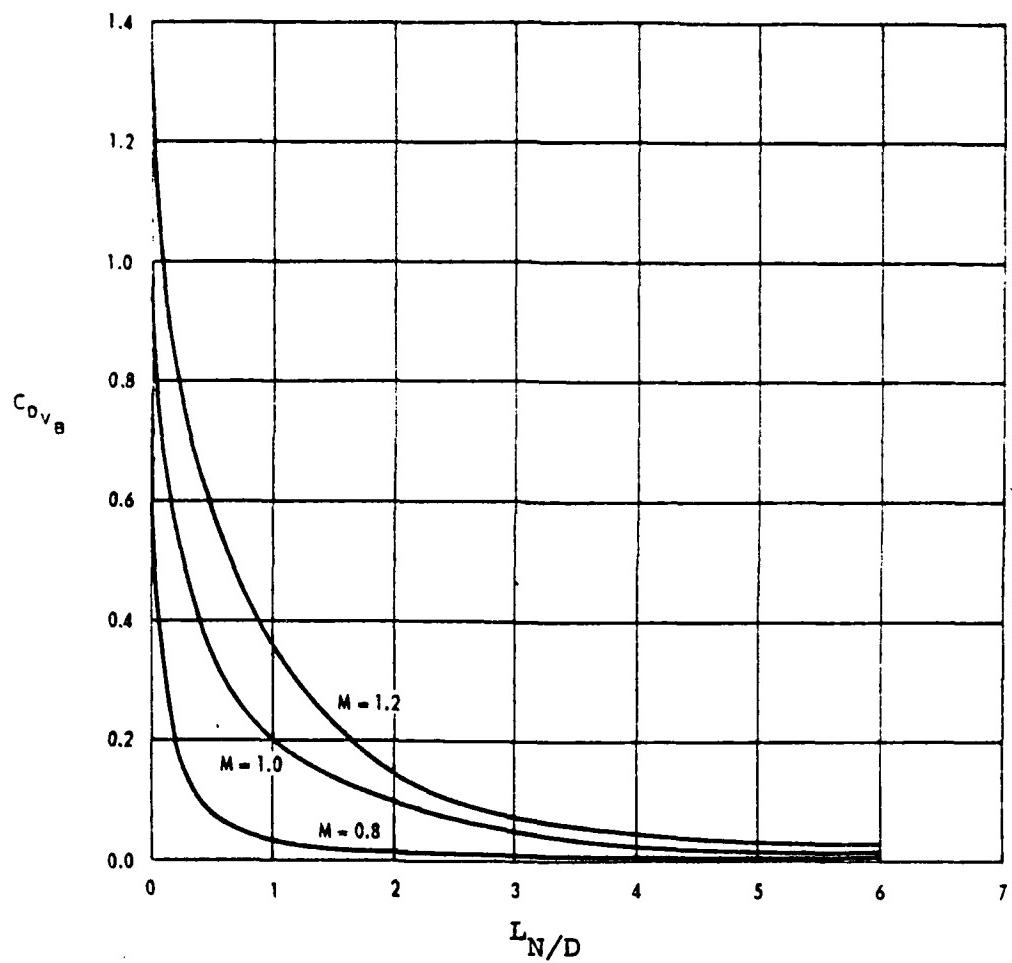


Figure (9-5). Transonic wave drag for ogival and blunted conical forebodies [18].

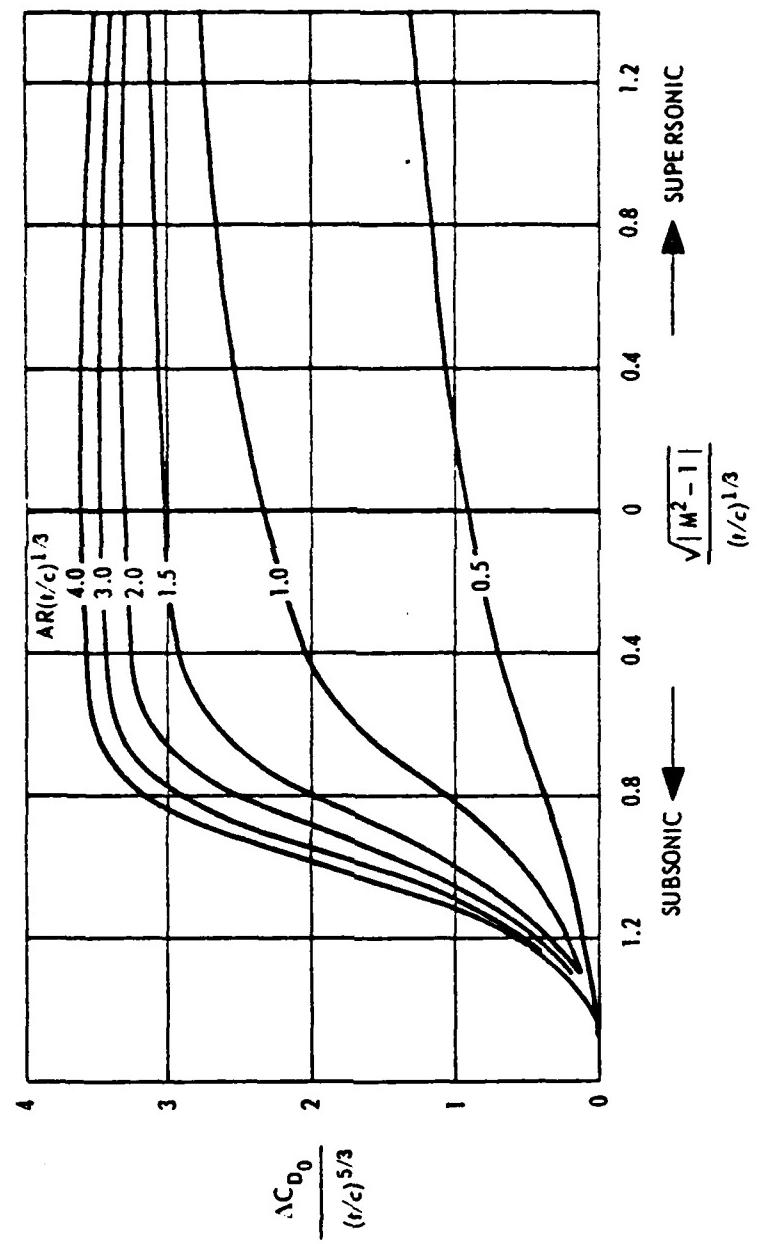


Figure (9-6). Transonic C_{D_0} for unswept wings [18].

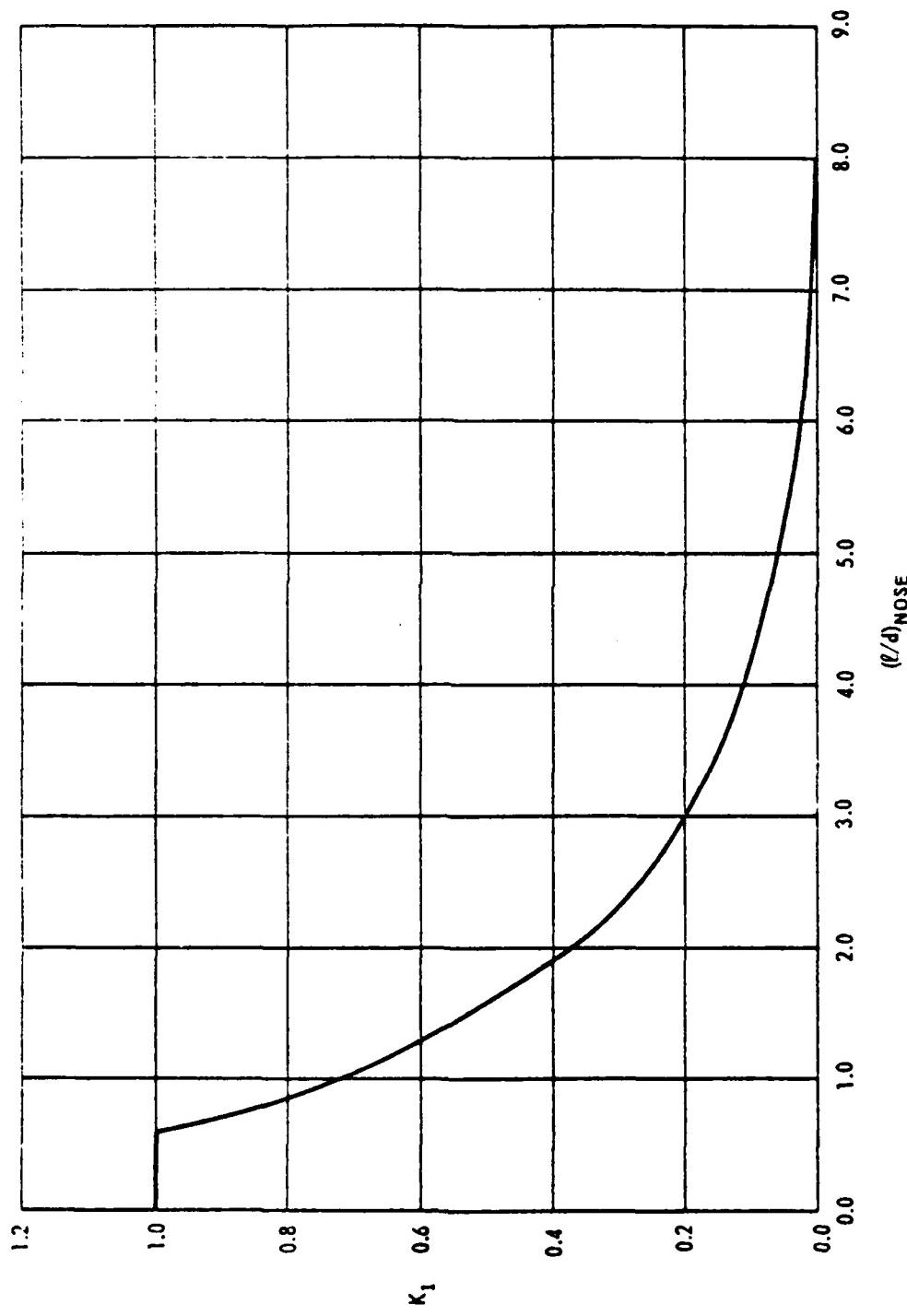


Figure (9-7). Ratio of C_D to C_{D_W} for various L/D to C_{D_W} hemispherical [18].

where

$$C_{D_0}' = (C_{D_0})_{M=1.2}$$

$$C_{D_0}'' = (C_{D_0})_{M=3.0}$$

C_{D_0}' is determined by using the methods for transonic flow.

The magnitude of the supersonic wave drag is highly dependent upon the nose shape. For this reason two methods are used to determine C_{D_0}'' .

Method I: This method is for blunted ogives, pointed ogives and blunted cones. In this method

$$\text{For } L_{N/D} \leq .5 \quad C_{D_0}'' = C_{D_0}'$$

$$\text{For } L_{N/D} \geq 8.0 \quad C_{D_0}'' = (C_{D_0})_{M=.8} + \Delta C_{D_{0W}}$$
$$+ \Delta C_{D_{0T}} + C_{A_B}$$

The above values are determined from the methods of transonic flow.

$$\text{For } L_{N/D} = .5 \text{ to } 8.0$$

$$C_{D_0}'' = K_1 C_{D_0}'$$

K_1 in the above equation is derived empirically and is given in Figure (9-7).

Method II: This method is for pointed conical noses. In this method C_{D_0}'' is determined by

$$C_{D_0}'' = (C_{D_0})_{M=0.8} + \Delta C_{D_{0W}} + \Delta C_{D_{0T}} + (C_{D_{VN}})_{M=3} + C_{A_B}$$

The first three terms in the above equation are found by transonic flow methods. The forebody wave drag, $C_{D_{VN}}$, for $M = 3.0$ is found using Figure (9-8). The nose semi-vertex angle, θ_N , is the same as σ in Chapter 6. In all flow regimes the base pressure drag is found as in Chapter 6.

2. Induced Drag

The induced drag due to angle of attack depends upon the flight regime the missile is in. For $M < .85$ and $AR > 3.0$ the induced drag is,

$$C_{D_i} = \frac{(C_L)^2}{\pi A Re}$$

where e is the Oswald efficiency factor = 0.7.

The tactical missile normally has an aspect ratio of less than 3.0. For all flight speeds with $AR < 3.0$ the induced drag is

$$C_{D_i} = C_L \tan \alpha$$

The component induced drag coefficients are found in the same manner as above using component lift coefficients.

3. Total Drag

From the zero-lift drag and induced drag the missile total drag becomes,

$$C_D = C_{D_0} + C_{D_i}$$

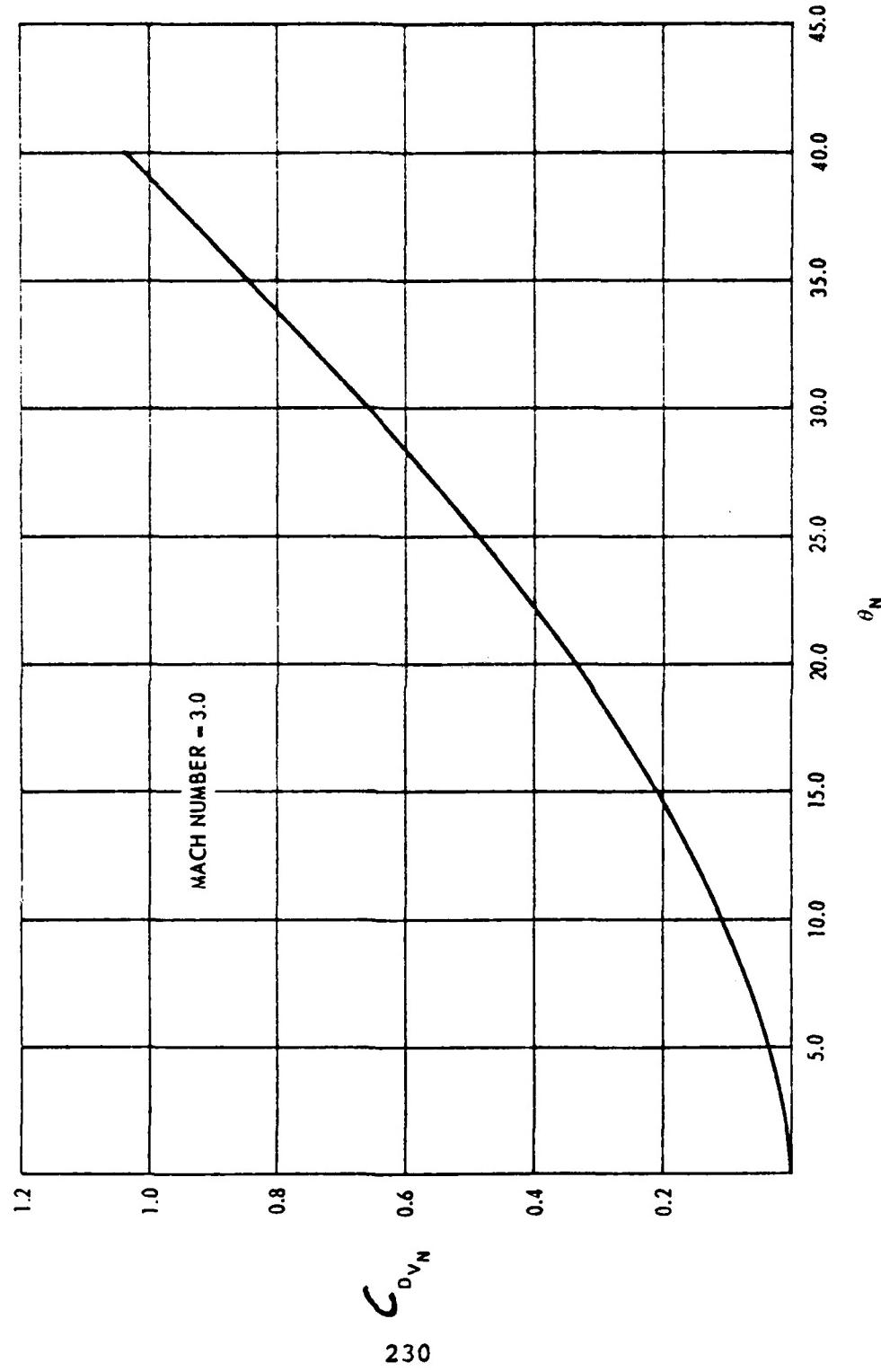


Figure (9-8). Wave drag of pointed conical nose [18].

D. PITCHING MOMENT CHARACTERISTICS

The pitching moment of a missile is equal to the sum of the pitching moments due to lift and drag forces acting on the body, wings and tails. If only small angles of attack are of interest, the pitching moment is due primarily to the lift forces. The methods presented here are for all angles of attack.

1. Body

The body alone pitching moment about its center of gravity is given from slender body theory and viscous cross-flow theory as

$$C_{M_B} = \left[\frac{V_B - S_B (L - x_{CG})}{S_{ref} L_{ref}} \right] \sin^2 \alpha \cos \frac{\alpha}{2} + n C_d c \left(\frac{S_p}{S_{ref}} \right) \left(\frac{x_{CG} - x_p}{L_{ref}} \right) \sin^2 \alpha$$

S_p = Planform area

x_p = Centroid of planform area

V_B = Body volume

S_B = Base area

With C_{M_B} given the center of pressure of the body becomes

$$(x_{c_p})_B = \left(\frac{x_{CG}}{L_{ref}} - \frac{C_{M_B}}{C_{N_B}} \right) L_{ref}$$

2. Wing (Fixed Surface)

The center of pressure location for the wing must be found before the pitching moment can be specified. The center of pressure of the various lift components are found by the method of Pitts, Nielsen and Maattari [13]. The lift

components are the same as in the lift section. The center of pressure of the lift on the wing in the presence of the body, $(\bar{x})_{W(B)}$, is found from Figure (9-10) if the flow is subsonic and Figure (9-11) if the flow is supersonic.

The center of pressure of the additional lift on the body in the presence of the wing, $(\bar{x})_{B(W)}$, is found using Figure (9-12) if the flow is subsonic. If the flow is supersonic the center of pressure is found from Figure (9-13) or Figure (9-14) depending on the parameters;

$$\beta AR(1 + \lambda)(1 + \frac{1}{m\beta})$$

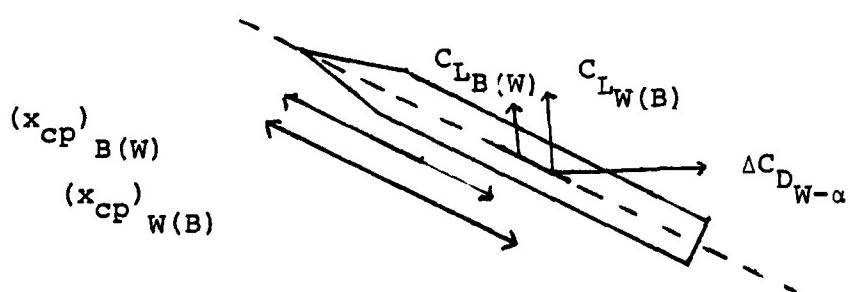
λ = taper ratio

$m = co + \Delta_{LE}$

If the center of pressure reference is moved to the nose, the following expressions result.

$$(x_{cp})_{W(B)} = (\frac{\bar{x}}{c_r})_{W(B)} (C_r)_W + x_w$$

$$(x_{cp})_{B(W)} = (\frac{\bar{x}}{c_r})_{B(W)} (C_r)_W + x_w$$



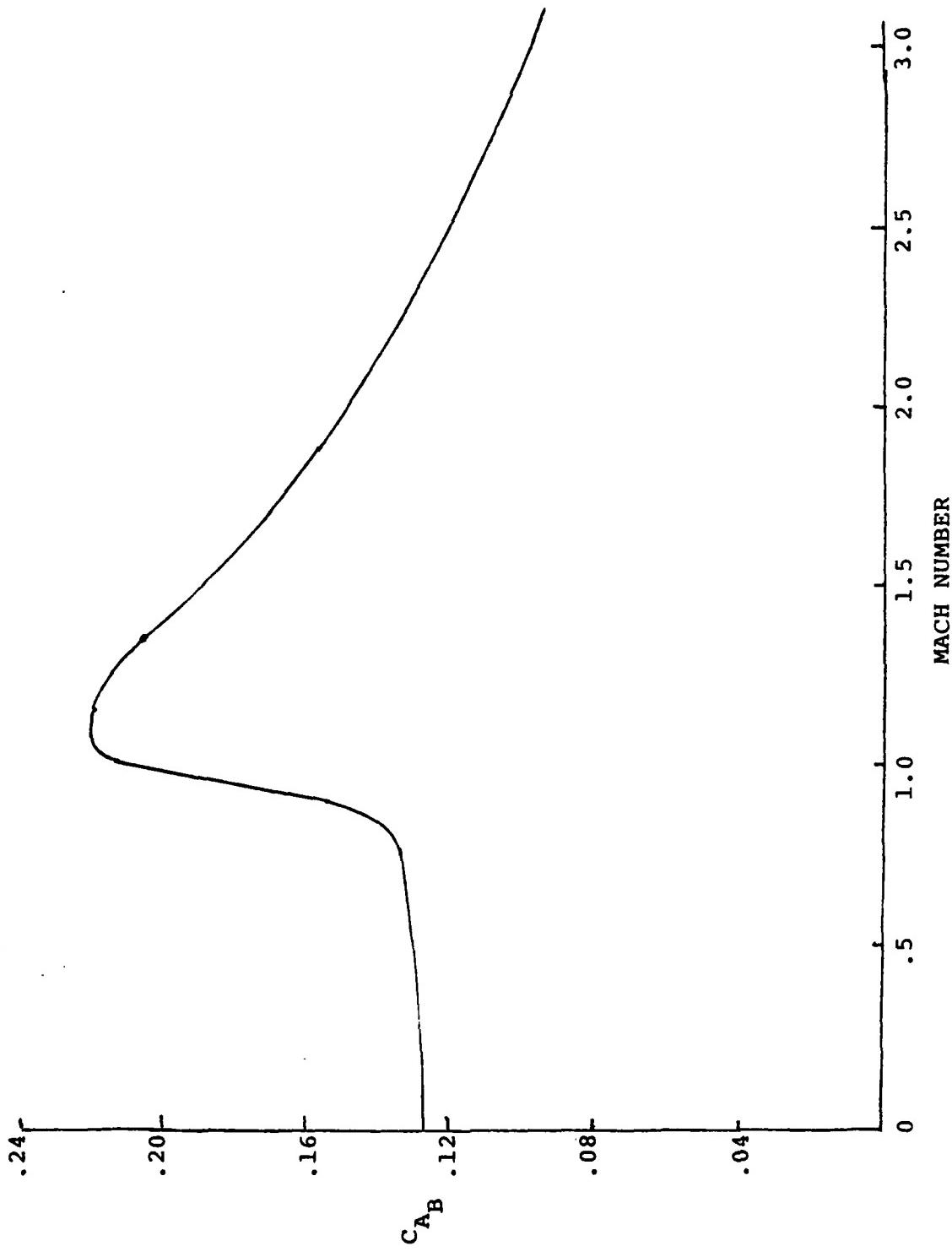


Figure (9-9). Mean base pressure curve.

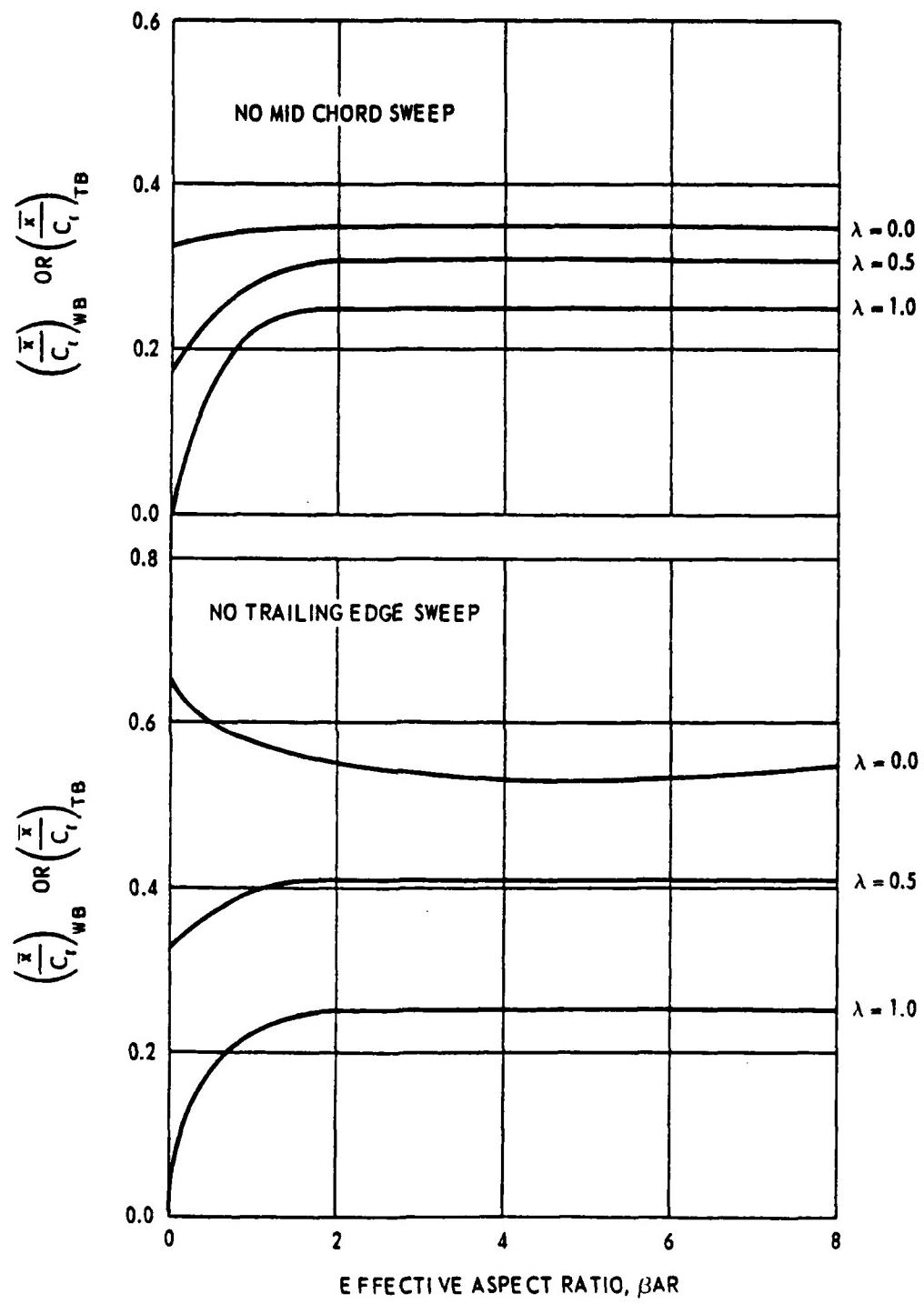


Figure (9-10). Wing center of pressure (subsonic) [18].

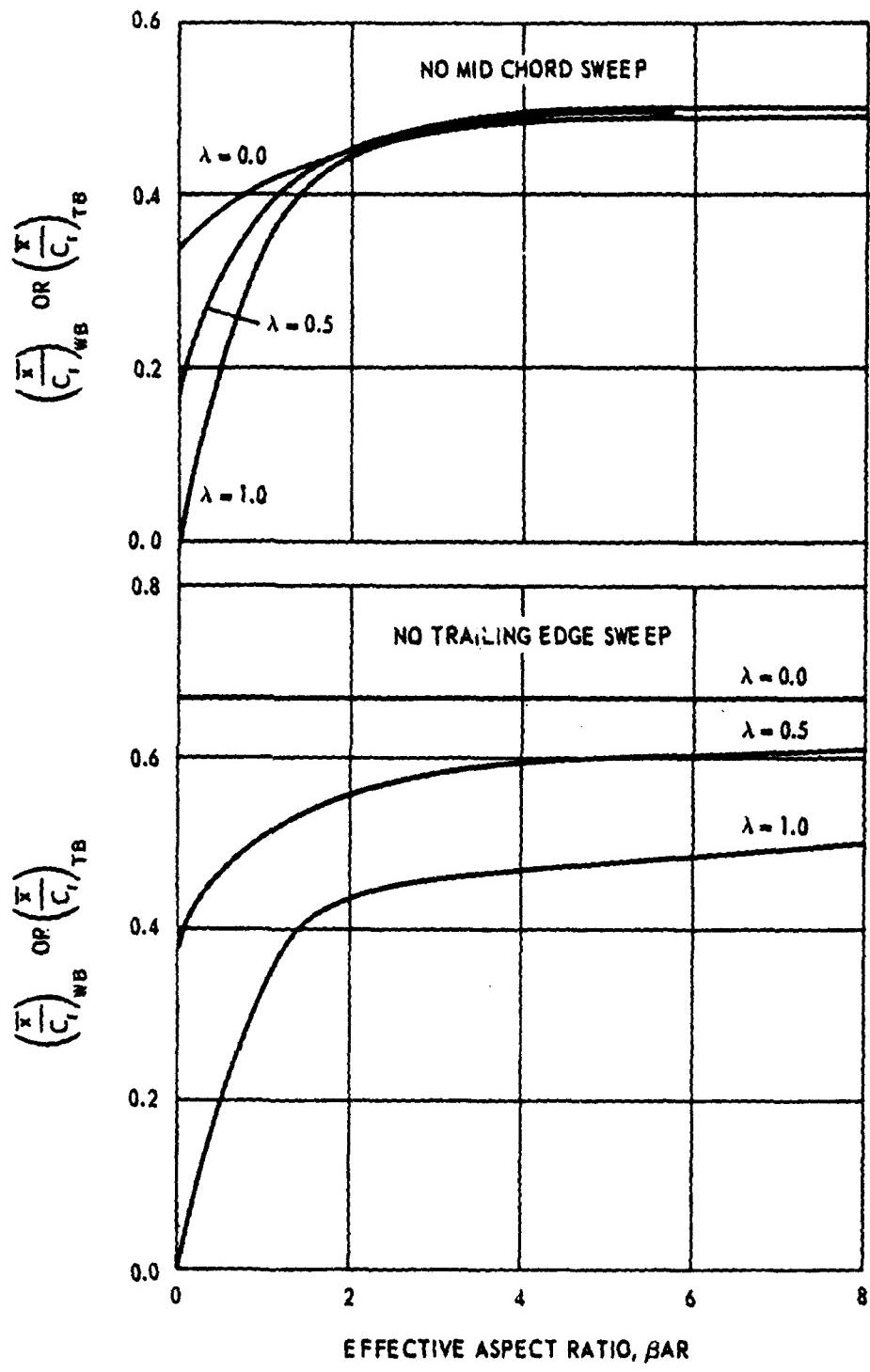


Figure (9-11). Wing center of pressure (supersonic) [18].

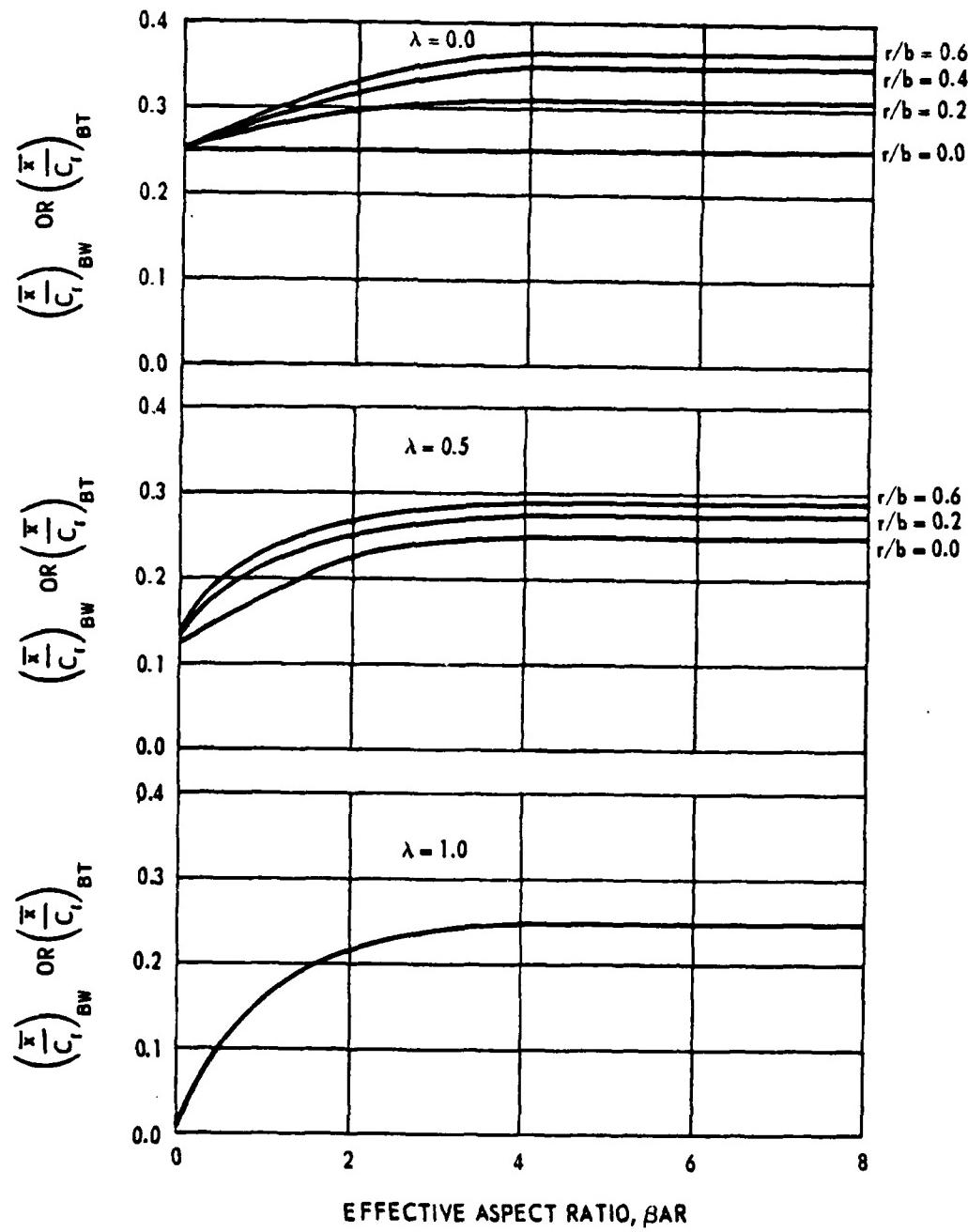


Figure (9-12). Body center of pressure [18].

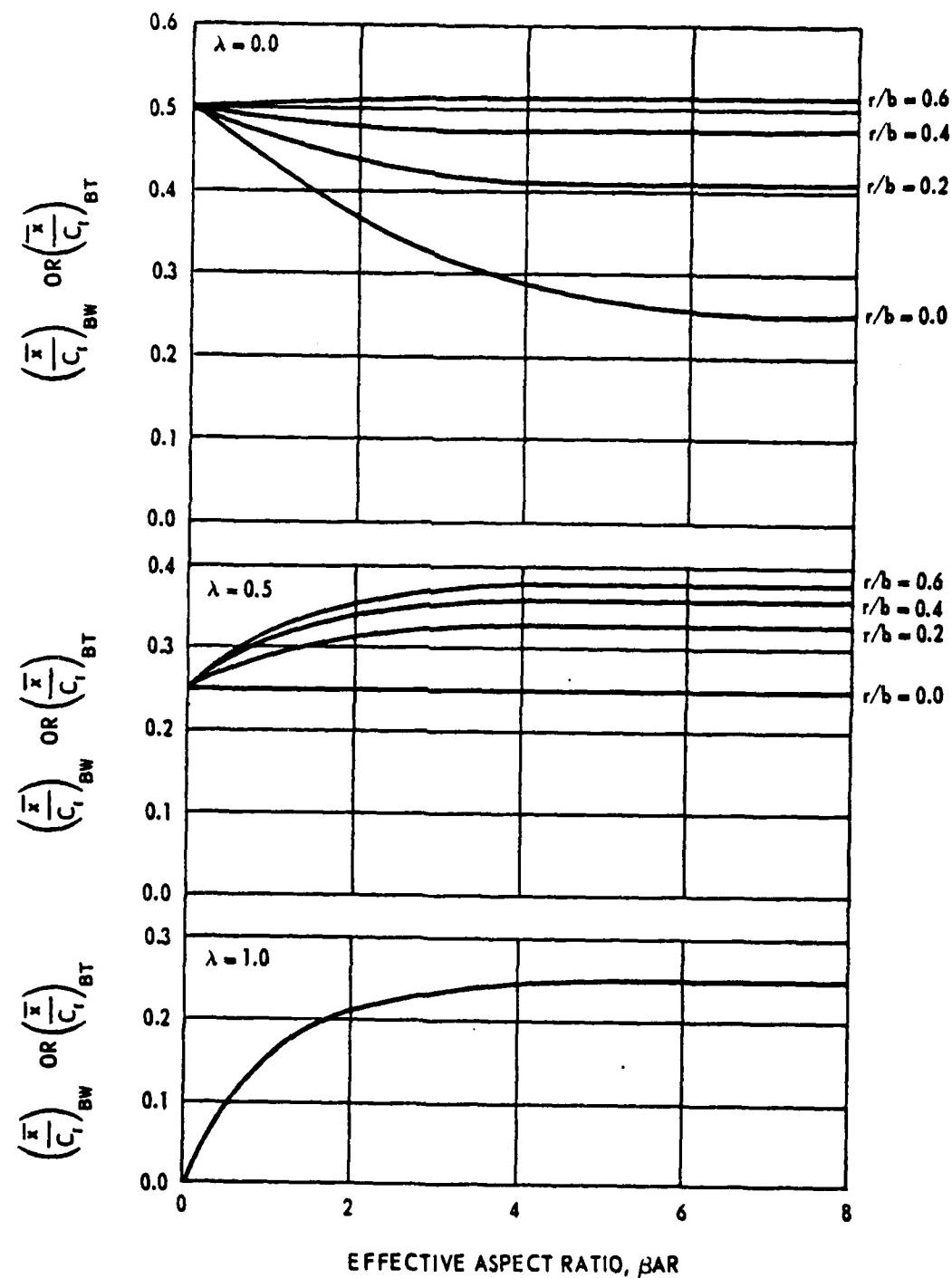


Figure (9-13). Body center of pressure [18].

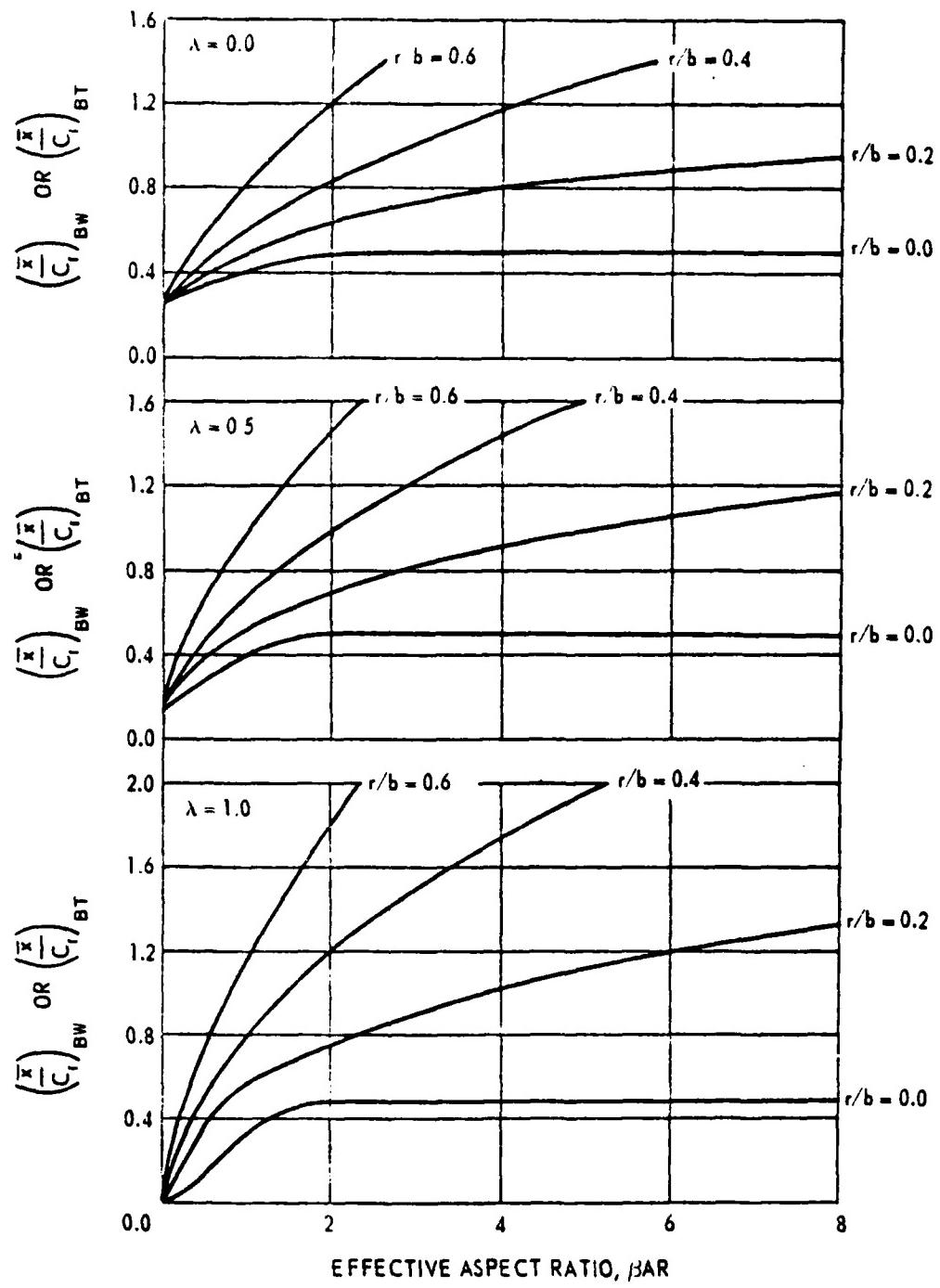


Figure (9-14). Body center of pressure [18].

From the above figure the entire pitching moment about the nose of the missile due to the forces acting on the wing is,

$$C_{M_W} = [(C_{L_{W(B)}} + C_{L_{W(V)}} + C_{L_i}) \cos\alpha + \Delta C_{D_{W-\alpha}} \sin\alpha] \left(\frac{(x_{cp})_{W(B)}}{L_{ref}} \right) \\ + [C_{L_{B(W)}} \cos\alpha] \left(\frac{(x_{cp})_{B(W)}}{L_{ref}} \right)$$

The viscous lift, $C_{L_{W(V)}}$ and lift-loss due to downwash, C_{L_i} , are not shown in the figure but are the same forces as found in the lift section. These forces are assumed to be acting through the center of pressure, $(x_{cp})_{W(B)}$.

An average center of pressure of the wing due to all forces acting on it can now be found as

$$(x_{cp})_W = \frac{C_{M_W} L_{ref}}{(C_{L_W} + C_{L_i}) \cos\alpha + \Delta C_{D_{W-\alpha}} \sin\alpha}$$

where

$$C_{L_W} = C_{L_{W(B)}} + C_{L_{B(W)}} + C_{L_{W(V)}}$$

3. Tail (Control Surface)

The tail pitching moment about the nose of the missile is found in the same way except now a control surface deflection must be included. The equations now become

$$\begin{aligned}
 C_{M_T} &= [(C_{L_{T(B)}} + C_{L_{T(V)}} + C_{L_i}) \cos \alpha + C_{L_{T(B)-\delta}} + \\
 &\quad (\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}}) \sin \alpha] \left(\frac{(x_{cp})_T}{L_{ref}} \right) \\
 &\quad + [C_{L_{B(T)}} \cos \alpha + C_{L_{B(T)-\delta}}] \left(\frac{(x_{cp})_{B(T)}}{L_{ref}} \right) \\
 (x_{cp})_T &= \frac{C_{M_T} L_{ref}}{(C_{L_{T-\alpha}} + C_{L_i}) \cos \alpha + C_{L_{T-\delta}} + (\Delta C_{D_{T-\alpha}} + \Delta C_{D_{T-\delta}}) \sin \alpha}
 \end{aligned}$$

where,

$$\begin{aligned}
 C_{L_{T-\alpha}} &= C_{L_{T(B)}} + C_{L_{B(T)}} + C_{L_{T(V)}} \\
 C_{L_{T-\delta}} &= C_{L_{T(B)-\delta}} + C_{L_{B(T)-\delta}}
 \end{aligned}$$

The wing and tail pitching moments above were found about the nose of the missile. Transferring the pitching moments about the center of gravity the complete missile pitching moment becomes

$$C_M = C_{M_B} + C_{M_W} \left[\frac{x_{CG} - (x_{cp})_W}{(x_{cp})_W} \right] + C_{M_T} \left[\frac{x_{CG} - (x_{cp})_T}{(x_{cp})_T} \right]$$

E. AEROL DESCRIPTION

It can be seen from the preceding description of the aerodynamics of a missile, that the process of obtaining a complete

description of the aerodynamic coefficients becomes an involved task. This was the justification for initially using linear aerodynamics. To complete an analysis a fast method of predicting the static aerodynamic characteristics is needed. The method needs to include both linear and nonlinear contributions as well as interference factors and must be applicable to all speed regimes. This analysis lends itself to programming on a digital computer. AEROL is a modification of the program AEROFC which was developed at the Naval Air Development Center by Mr. F.A. Kuster, Jr. [17]. The program is essentially as written except for calculation of planform areas and centroids. The program was also modified for use on the Naval Postgraduate School CP/CMS system.

AEROL consists of a main program and three subroutines. The inputs to the program are the geometric characteristics of the missile, the flight conditions, engine and inlet type and the protuberance drag. The output consists of the aerodynamic coefficients for lift, drag and moment. The component contribution to these coefficients are also given. The component and overall center of pressure are also determined.

Subroutine GEOSUB; This subroutine calculates the missile wetted area and the Reynolds number per foot based on the flight altitude.

Subroutine CLASUB; This subroutine calculates the aerodynamic surface lift-curve slopes.

Subroutine CATSUB; This subroutine calculates center of pressure locations, cross-flow drag coefficients, and interference factors.

Main Program; The main program assumes the control surface is the tail. This is regardless of the method of control (Wing, Tail, Canard). Because of this, care must be taken to input the right data for the tail. For instance, if the missile is a wing control missile, the wing data is input as the tail and the tail data as the wing. Figure (9-15) and Table (9-I) give a complete listing of the input data. Table (9-II) is a list of the output data. Appendix A is a listing of the program as modified for use on the Naval Postgraduate School IBM 360 computer.

1. Verification of AEROL

Before using the program an attempt was made to verify its prediction techniques and find any pitfalls in its use. To accomplish this the program was run for various configurations for which experimental data were available, and the results were compared. The comparisons are shown in Figures (9-16) to (9-25) from references [19] - [22] which are NASA technical notes and memoranda which report results of wind tunnel tests for various body-wing-tail combinations. In Figures (9-16) to (9-25) the solid lines are AEROL predictions.

a. NASA TN D-6996

This technical note presents a method of predicting aerodynamic characteristics for bodies alone at angles of attack from 0 to 180 degrees. Several nose-body combinations are given.

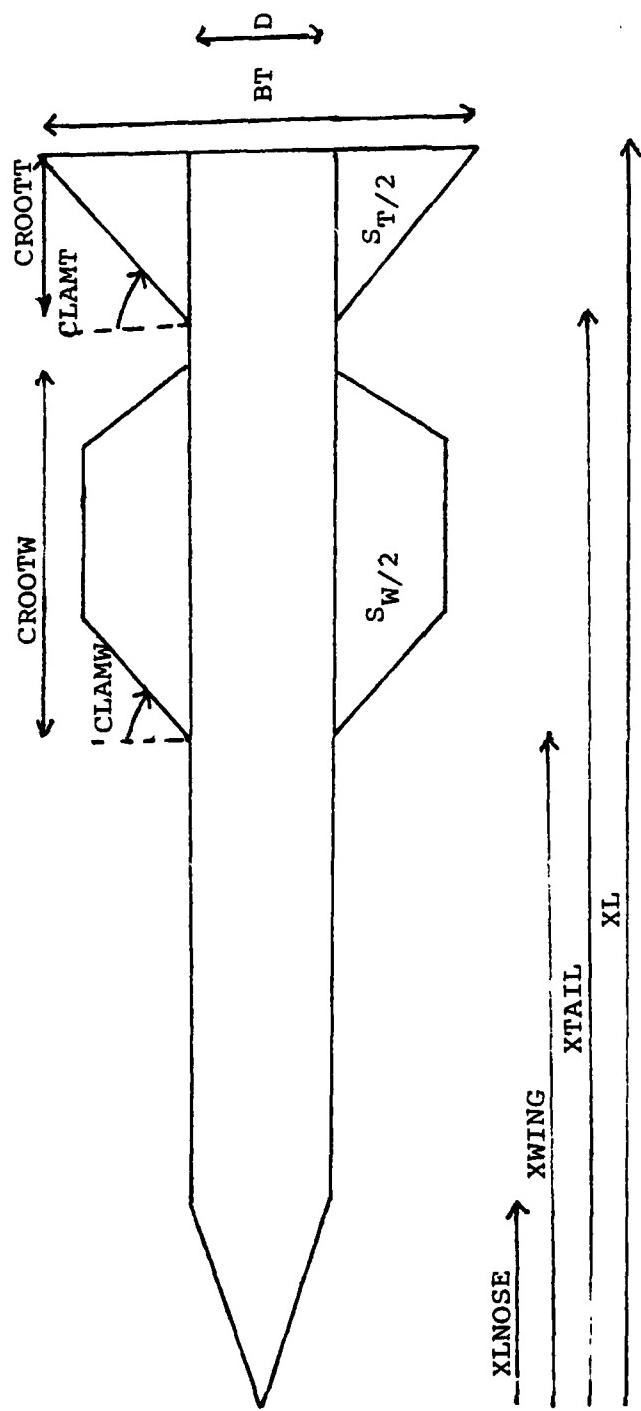


Figure (9-15). AERO1 input data.

TABLE (9-I)

Input Data

<u>Variable</u>	<u>Name</u>	<u>Format</u>	<u>Meaning</u>
1	ICSC	12	Type of control (wing, tail, canard)
2	INOSE	12	Nose shape (ellipsoidal, cone, ogive)
3	IDT	12	Number of control surface deflections
4	IM	12	Numer of Mach numbers
5	IAL	12	Number of angles of attack
6	NBODY	12	Number of configurations
7	ISWPW	12	Wing shape (delta,nondelta)
8	IAFBW	12	Missile body after wing
9	IWEPW	12	Leading edge sweep indicator
10	NWING	12	Number of wings
11	ISWPY	12	Tail shape (delta,nondelta)
12	IAFBT	12	Missile body after tail
13	ISWEPT	12	Leading edge sweep indicator
14	NTAIL	12	Number of tails
15	XLAMW	F10.5	Wing taper ratio
16	CLAMW	F10.5	Wing leading edge sweep
17	BW	F10.5	Wing span
18	CROOTW	F10.5	Wing root chard
19	SW	F10.5	Wing exposed area
20	XMACW	F10.5	Wing mean geometric chord
21	XWING	F10.5	Distance to wing leading edge

<u>Variable</u>	<u>Name</u>	<u>Format</u>	<u>Meaning</u>
22	TOVCW	F10.5	Wing thickness to chord ratio
23	XLAMT	F10.5	Tail taper ratio
24	CLAMT	F10.5	Tail leading edge sweep
25	BT	F10.5	Tail span
26	CROOTT	F10.5	Tail root chord
27	ST	F10.5	Tail exposed area
28	XMACT	F10.5	Tail mean geometric chord
29	XTAIL	F10.5	Distance to tail leading edge
30	TOVCT	F10.5	Tail thickness to chord ratio
31	HT	F10.5	Altitude
32	D	F10.5	body diameter
33	XL	F10.5	Body length
34	XLNOSE	F10.5	Nose length
35	XCG	F10.5	Center of gravity location
36	AREA	F10.5	Reference area
37	XREF	F10.5	Reference length
38	ENGINE	F10.5	Engine code
39	INLET	F10.5	Inlet code
40	BETA	F10.5	Boattail angle
41	DBASE	F10.5	Base diameter
42	DJET	F10.5	Nozzle exit diameter
43	XLABOD	F10.5	Boattail length
44	CDPROT	F10.5	Protuberance drag coefficient

TABLE (9-II)

Output Data

<u>Variable Name</u>	<u>Meaning</u>
AL	Angle of attack
CLTOT	Total coefficient of lift
CDTOT	Total coefficient of drag
CLWP	Wing panel coefficient of lift
CLBW	Additional lift on body due to wing coefficient
CLTP	Tail panel coefficient of lift
CLBT	Additional lift on body due to tail coefficient
CLB	Body alone lift coefficient
CDI	Induced drag coefficient
CNWP	Wing panel normal force coefficient
CNTP	Tail panel normal force coefficient
CLTD	Coefficient of lift due to tail deflection
CDTD	Coefficient of drag due to tail deflection
CN	Total normal force coefficient
CA	Total axial force coefficient
XCPW	Wing center of pressure
XCPT	Tail center of pressure
XCP	Total missile center of pressure
CM	Total pitching moment about center of gravity

Figures (9-16) to (9-18) compare the coefficients predicted by AEROL with those obtained for body number 9 in the NASA report. The normal force coefficient is predicted well throughout the range. The moment and axial force trends are predicted by the program but large errors exist throughout the range of angles of attack.

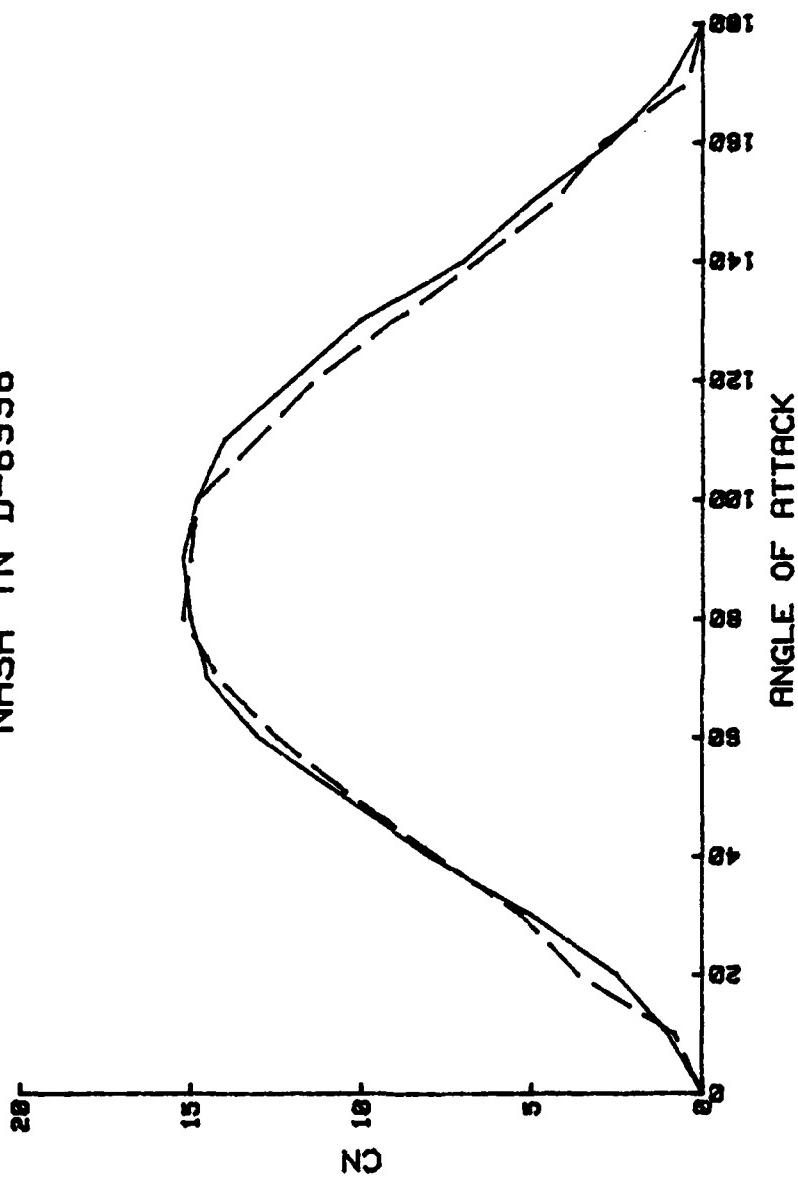
b. NASA TM X-2367

This technical memorandum investigates the aerodynamic characteristics of various cruciform body-wing combinations. The coefficients for this configuration agree very well with experimental values up to 10 degrees angle of attack. Beyond this value the lift and moment coefficients predicted by AEROL exceed the experimental values by as much as 10 percent. Although the exact cause of this error was not investigated, it may be partially explained by the nose shape of the body. The nose is a combination ogive and cone. For purposes of AEROL it was assumed an ogive. The greater presented area of the ogive would contribute to a higher C_L and C_m through the cross-flow terms in these coefficients. Figures (9-19) through (9-21) compare the predicted with the experimentally determined coefficients for the wing-tail configuration of this reference at $M = .9$.

c. NASA TM X-2780 and NASA TM X-2289

These technical memoranda investigate the aerodynamics of a delta wing missile using tail control and a tail-less cruciform missile. As shown in Figures (9-22) to (9-25) there is excellent agreement between the experimental values of the aerodynamic coefficients and those predicted by AEROL.

NASA TN D-6996



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Figure (9-16). Body alone normal force.

NASA TN D-6996

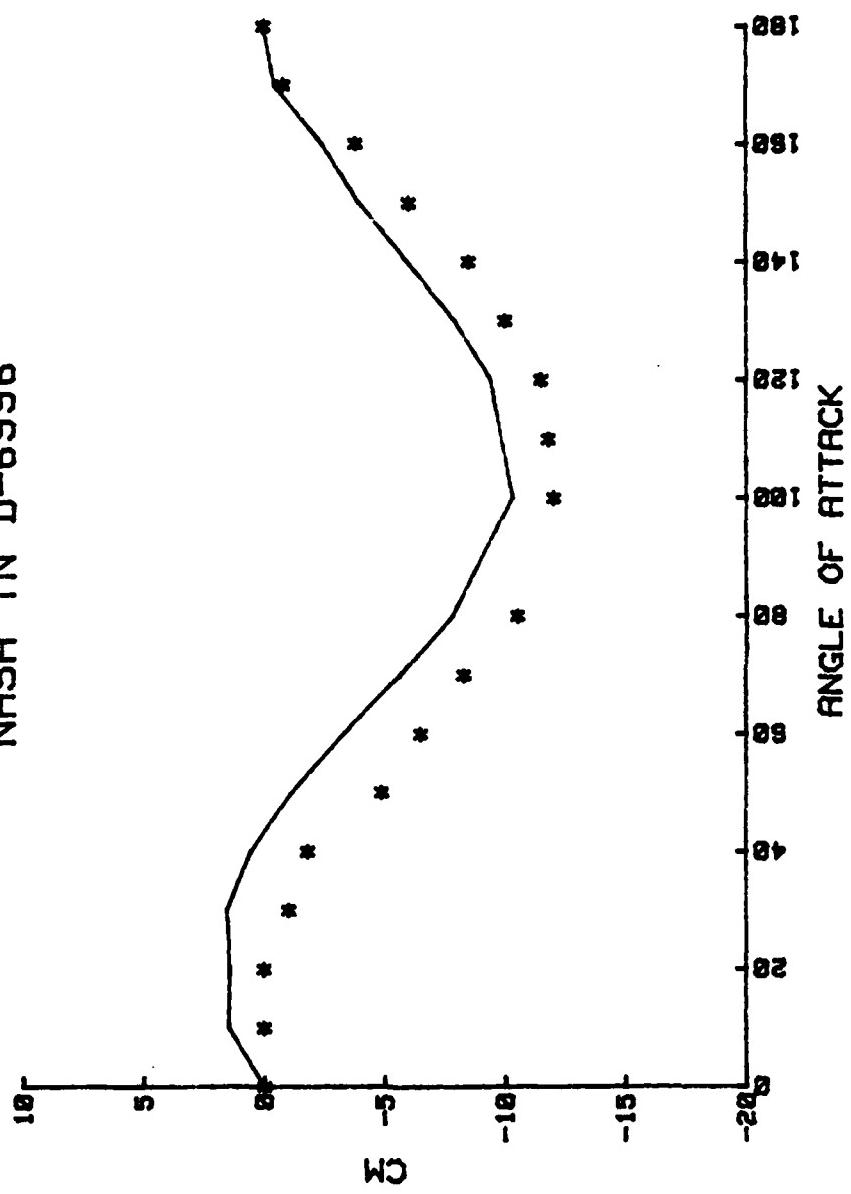


Figure (9-17). Body alone moment.

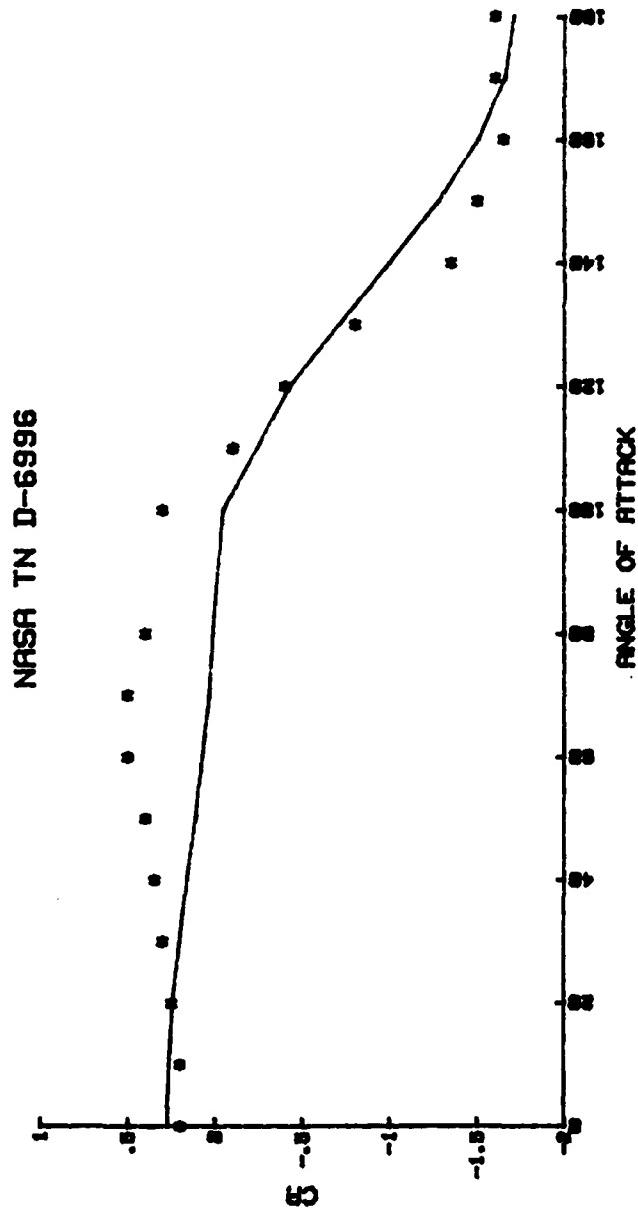


Figure (9-18). Body alone axial force.

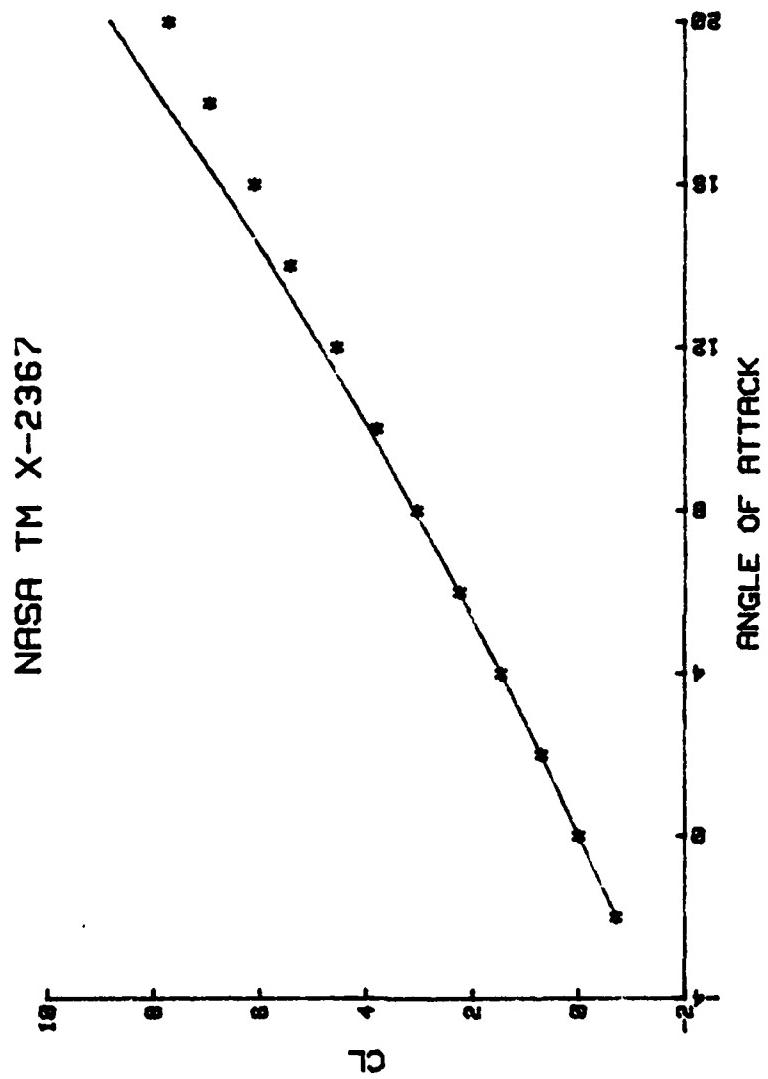


Figure (9-19). Tail-body lift force.

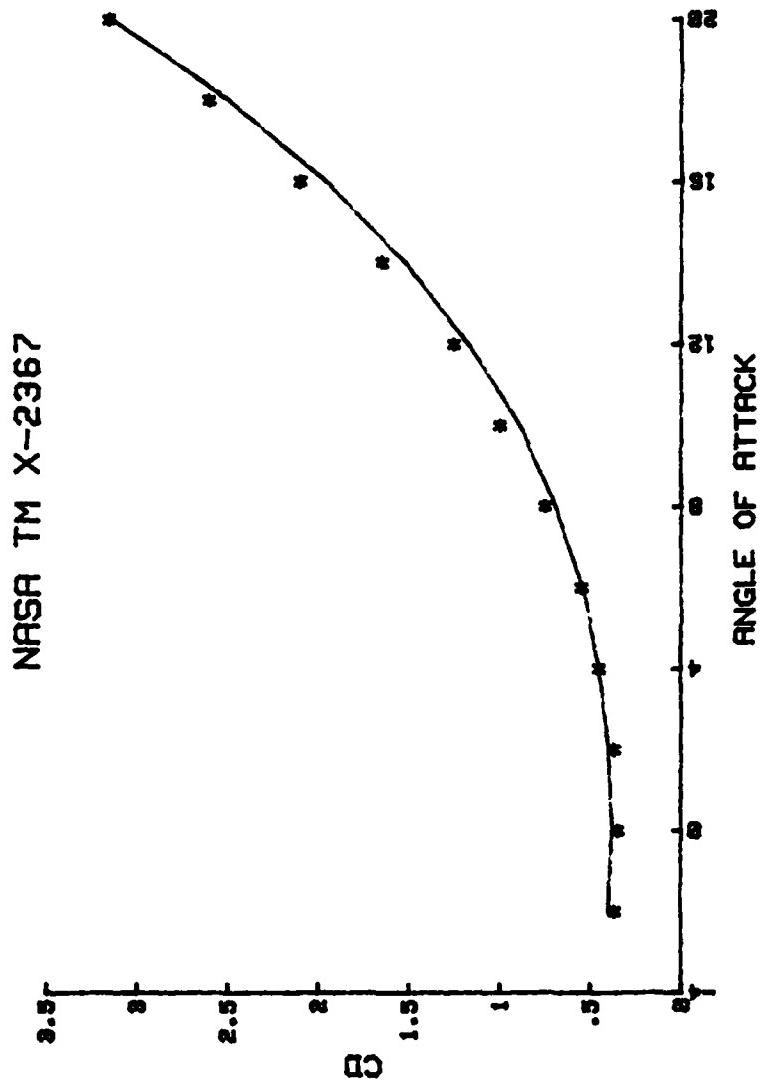


Figure (9-20). Tail-body drag force.

NASA TM X-2367

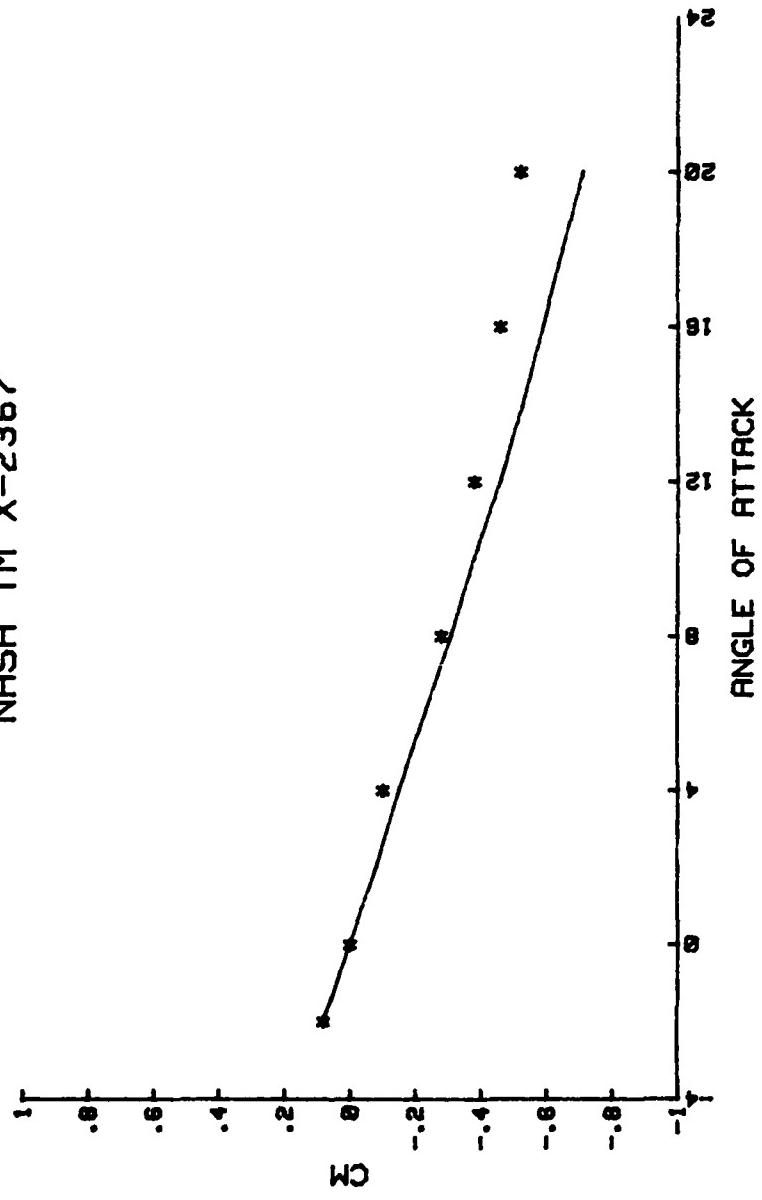


Figure (9-21). Tail-body moment.

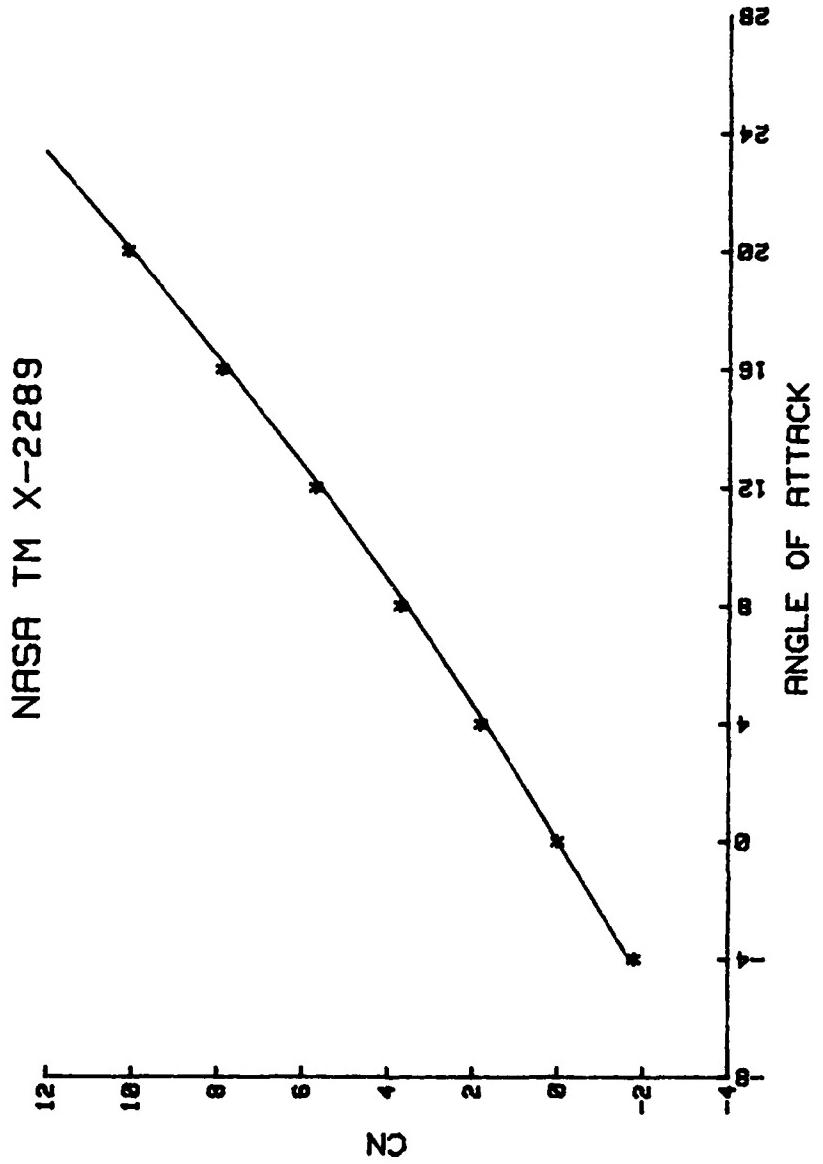


Figure (9-22). Wing-body normal force.

NASA TM X-2289

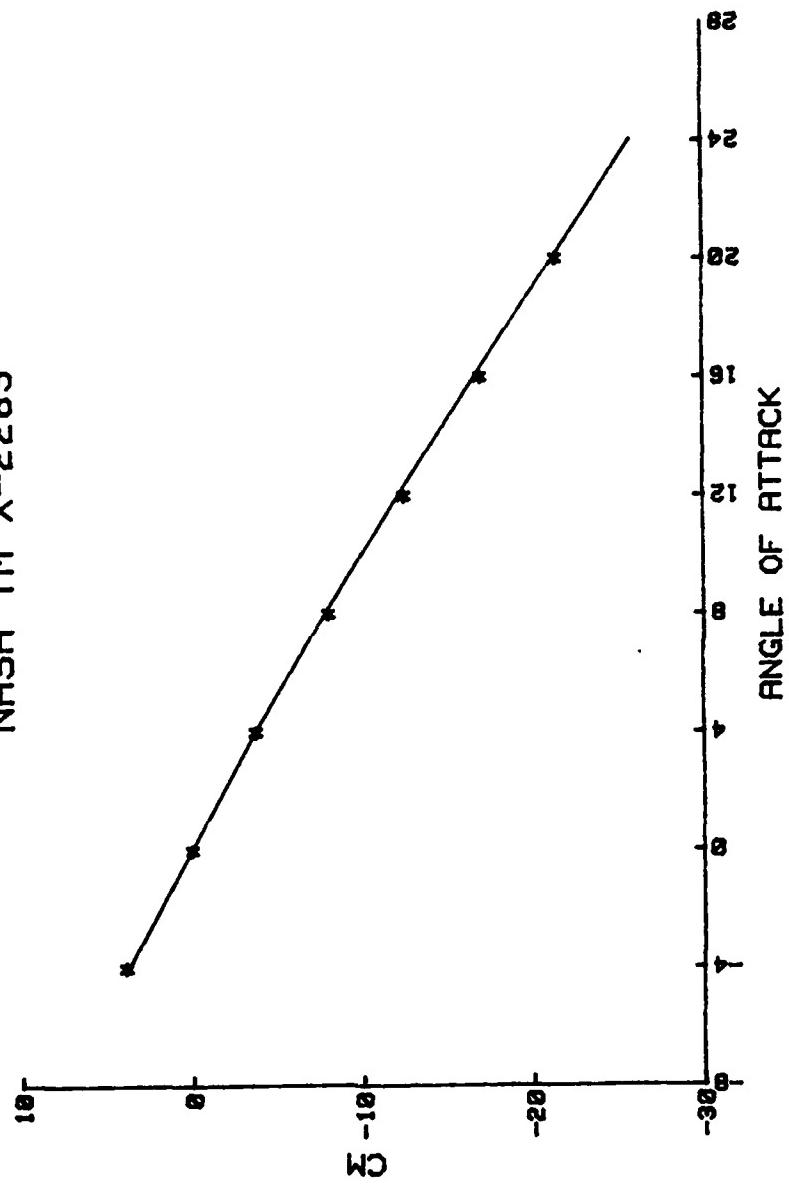
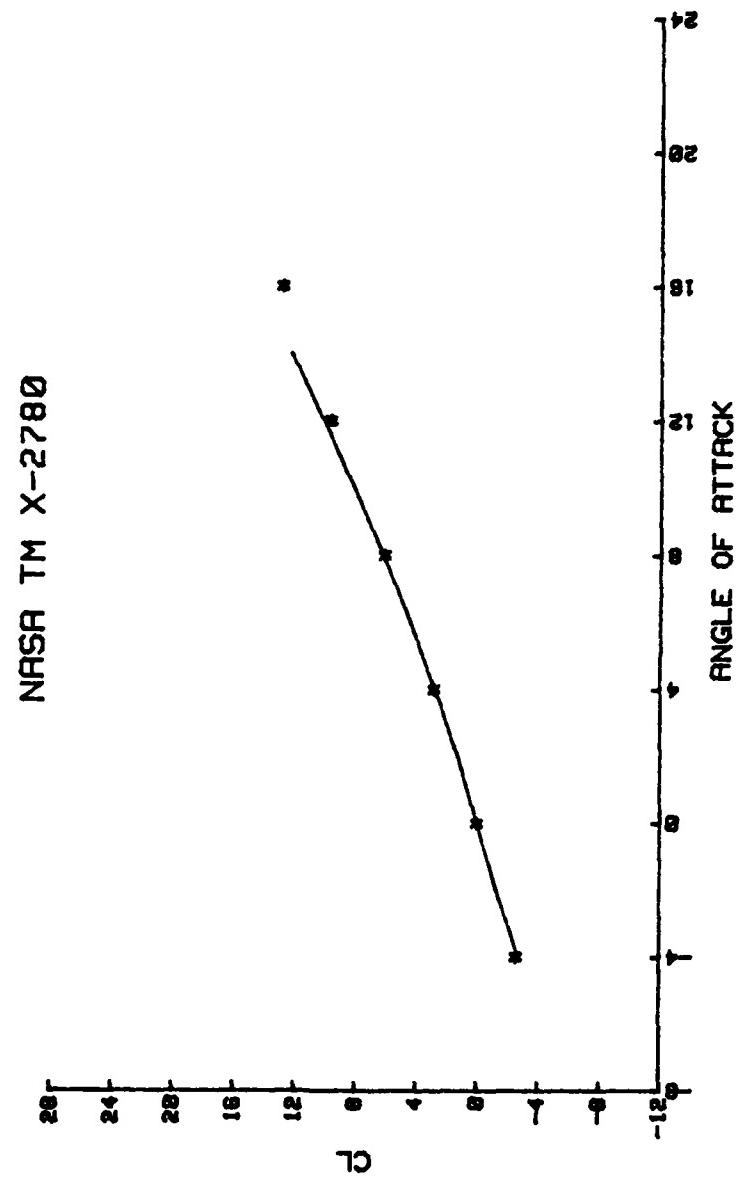


Figure (9-23). Wing-body moment.

Figure (9-24). Complete missile lift force.



NASA TM X-2780

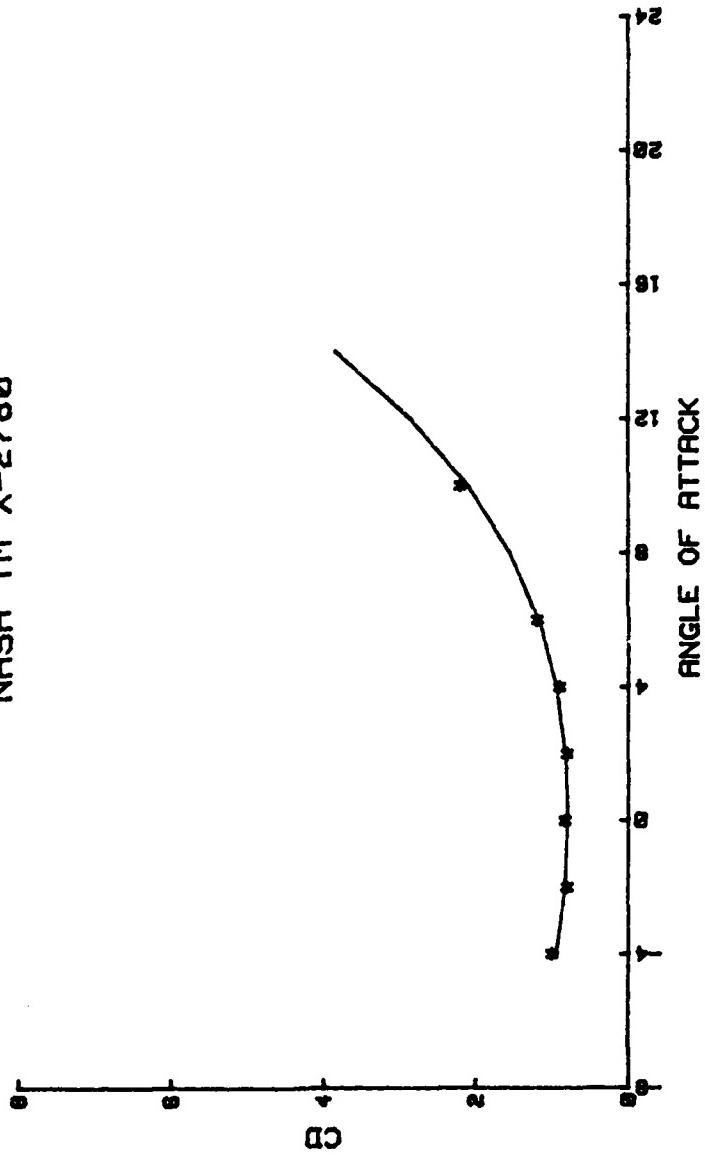


Figure (9-25). Complete missile drag.

F. COMPONENT WEIGHTS

As with the gross weight, the component weights can be determined through the use of parametric regression equations. If the dimensions of the components are known, the following formulas can be used to determine the component weights [7].

Aero Surface Weight

The weight of one wing panel is given by,

$$w_{AS} = 6.77483 (E_{AS})^{1.02} (AR_{AS})^{.56} \quad (1)$$

E_{AS} Exposed area of one panel, ft^2

AR_{AS} Aspect ratio of one panel

Body Structure Weight

$$w_{BS} = .0604 (L_{BS})^{.64} (D_{BS})^{1.77} \quad (2)$$

L_{BS} Length of body to be covered by structure. This does not include the rocket motor unless a separate structure surrounds the motor casing. (inches)

D_{BS} Diameter of body structure (inches)

Internal Systems Weight

$$w_{IS} = .00485 (W_G)^{.74} (L_{IS})^{1.00} (D_{IS})^{.42} \quad (3)$$

W_G Gross weight of the missile

L_{IS} Length of subsystem (inches)

D_{IS} Diameter of missile (inches)

In the above equation, the internal system includes guidance, avionics and control.

1. Design Example (Component Sizing)

If the guidance and the control sections are kept at 10 percent of the total length, and the remaining components are as sized previously, the design is as shown in Figure (9-27). Since these components are considered internal systems, their weights can be determined from equation (3).

$$W_{cont} = .00485 (1809)^{-79} (.1L) (12.5)^{.42}$$

$$W_{cont} = W_{guid} = 83.64 \text{ lb}$$

The wing and canard weights are found using equation (1) and Figure (9-26).

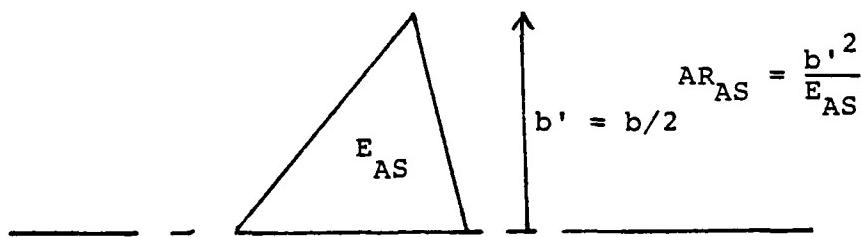


Figure (9-26). Aero surface weight.

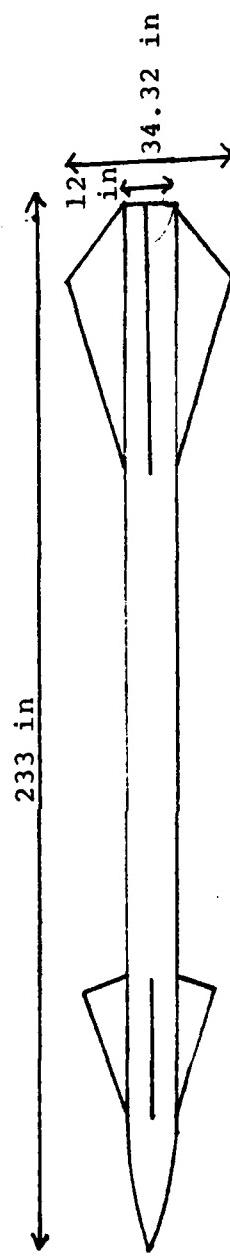


Figure (9-27). Final design.

The wing weight is four times the weight of one panel.

$$W_W = 43.42 \text{ lb}$$

Similarly the canard weight is

$$W_C = 9.88 \text{ lb}$$

If the entire body is covered by structure, the body structure weight becomes

$$W_{st} = .0604 (232)^{.64} (12.5)^{1.77}$$

$$W_{st} = 172.38 \text{ lb}$$

The following component table can now be constructed.

<u>Component</u>	<u>Length (in)</u>	<u>Weight (lbs)</u>	<u>Center of Gravity (in)</u>
Control	23.2	83.64	39.48
Guidance	23.2	83.64	62.68
Warhead	31.25	28.26	89.91
Sustainer	87.51	618.92	149.29
Booster	26.31	159.46	206.20
Motor Casing	113.82	59.50	162.44
Wings		43.42	211.40
Canards		9.88	44.08
Structure	232.00	172.38	116.00

The complete missile weight is then 1264.62 lbs.

The center of gravity from the above table is

$$x_{CG} = 139.44 \text{ in}$$

The center of gravity at the end of boost is

$$(x_{CG}) = 129.85 \text{ in}$$

G. DESIGN EXAMPLE (FINAL ANALYSIS)

From the design parameters so far, a complete description of the missile can be determined. Figure (9-27) is a drawing of the missile. The launch conditions for the missile were specified as,

$$M_M = 1.5$$

$$h = 10,000 \text{ ft}$$

$$W_G = 1264.62$$

$$x_{CG} = 139.44 \text{ in}$$

From these conditions the input data for AERO1 is shown in Table (9-III). The format is the same as that of the output of AERO1 and is printed as a check to ensure that the input data was entered properly. The output is shown in Table (9-IV).

Figure (9-28) is a plot of the coefficient of moment versus angle of attack for the launch condition and for the beginning of cruise. The center of gravity has moved forward approximately 10 inches for the beginning of cruise so that the stability has

TABLE (9-III). AERO1 Input

Aero Launch Conditions

ICSC 3	INOSE 2	INDELT 1	INMACH 1	INALPH 10	INBODY 1	CROOTW	SW	XHACM	XHARO
ISWPW	IAFBW	XLAMW	CLAMW	BW					
2	0	0.0		2.860000	2.860000	5.049999	4.700000	3.370000	14.370000
ISHPW2	IAFBW2	XLAMW2	CLAMW2	BW2	CROW2	SU2	XHACW2	XHIN2	XHIN2
0	0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
ISWPT	IAFBT	XLANT	CLANT	BT	CROTT	ST	XHACT	XTAIL	XTAIL
2	1	0.0	75.500000	2.360000	2.750000	1.870000	1.830000	2.557971	2.557971
MT	10000.000	0	XLNOSE	XRCG	ANFA	XREF			
		1.000	19.420	2.320	11.620	0.785	1.000		
TOVCU	TOVCU2	TOVCT	ISWEPT	ISWEPT	ISWEP2	NHMG	HTAIL	HEMING2	
0.030	-0.060	0.030	0	0	0	4	4	0	
ENGINE	ENSLT	BETA	DWASE	DJET	XLUJOD	CDPROT			
1.000	0.0	0.0	1.000	0.852	0.0	0.0			

TABLE (9-IV). AERO1 Output

CNDI NL=0.0 CNDI WF=0.0 CNDI PROT=0.0
CNDI SC=0.0335 CNDI F=0.0736 CNDI T=0.0320 CNDI SC=0.0292 CNDI WF=0.4091

VXM= 3.00 DELTA= 0.0

	AL	CLTOT	CDTUT	CLWP	CLBW	CLTP	CLBT	CLB	CDI	CNWIP	CNTP	CLTD	CDTD	CN	CA	XCPW	XCP	XCP	CM	
-2.	-0.53	0.38	-0.13	-0.07	-0.14	-0.05	-0.09	0.01	-0.18	-0.14	0.00	0.00	-0.55	0.36	17.78	4.42	3.38	10.50	-0.17	
0.	0.00	0.36	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.36	17.78	4.42	-0.00	8.42	0.00
2.	0.53	0.38	0.18	0.07	0.14	0.05	0.09	0.01	0.18	0.14	0.00	0.00	0.55	0.36	17.78	4.42	3.38	10.50	0.17	
4.	1.24	0.45	0.45	0.15	0.29	0.10	0.25	0.08	0.46	0.29	0.00	0.00	1.27	0.36	17.78	4.42	5.10	11.00	-0.23	
6.	2.06	0.59	0.77	0.22	0.44	0.15	0.49	0.23	0.77	0.44	0.00	0.00	2.10	0.37	17.78	4.42	6.17	11.20	-0.79	
8.	3.00	0.86	1.10	0.29	0.60	0.20	0.81	0.49	1.11	0.60	0.00	0.00	3.05	0.43	17.78	4.42	6.92	11.28	-1.41	
10.	4.05	1.28	1.45	0.37	0.76	0.25	1.23	0.91	1.46	0.76	0.00	0.00	4.12	0.55	17.78	4.42	7.48	11.32	-2.04	
12.	5.24	1.91	1.82	0.43	0.93	0.29	1.77	1.55	1.82	0.92	0.00	0.00	5.32	0.78	17.78	4.42	7.93	11.32	-2.64	
14.	6.60	2.34	2.20	0.50	1.09	0.34	2.47	2.48	2.20	1.09	0.00	0.00	6.71	1.16	17.78	4.42	8.29	11.29	-3.16	
16.	8.96	5.02	2.58	0.57	1.26	0.38	4.18	4.66	2.57	1.25	0.00	0.00	9.15	2.35	17.78	4.42	8.86	11.13	-2.85	

CN1:IL=0.0 CNRAFT=0.0 CNPROT=0.0

CN03=0.2688 CNOW=0.0664 CNOT=0.0289 CNHIS=0.0301 CNMBT=0.3642

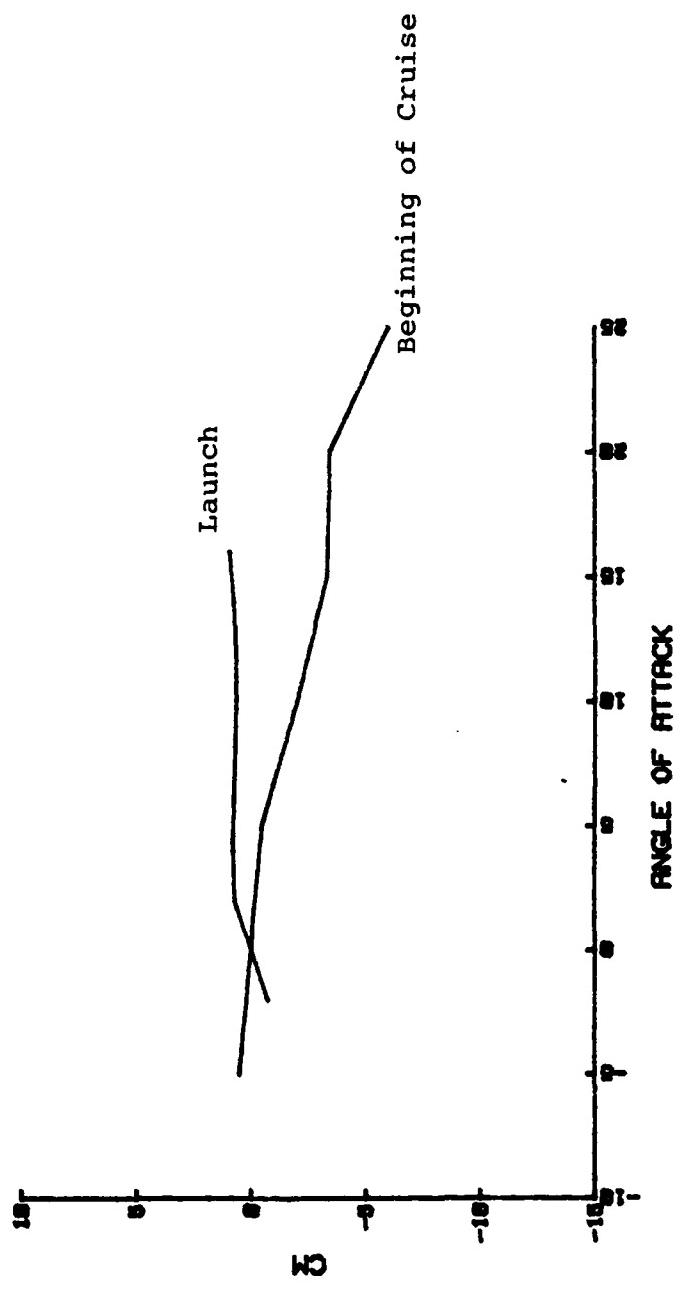


Figure (9-28). Static stability.

increased as indicated in this figure. The performance objective from the beginning of the design was to provide a 31 g maneuver capability. Table (9- V) shows the output of AERO1 for Mach numbers of 1.5 to 3.0 and control deflections of 0.0 to 30 degrees. From this output the trimmed normal force can be found and the corresponding load factor is then,

$$n = \frac{C_{N_{TR}} q S_{ref}}{W}$$

The following values of maneuver load factor were found using AERO1.

M	$C_{N_{TR}}$	$q(\text{lb}/\text{ft}^2)$	n(g's)	δ req
2.5	18.80	6369	85.00	10 deg.
2.0	24.50	4076	70.99	10 deg.
1.5	12.00	2293	19.56	10 deg.

As indicated in the above table, the missile meets the maneuvering specifications of the operational requirements.

TABLE (9-V). AEROL Output

VXM= 1.50 DELTA= 0.0

	AL	CLTOT	CNTOT	CLMP	CLBM	CLTP	CLBT	CLB	CDI	CNWP	CNTP	CLTD	CDTD	CN	CA	XCPW	XCP	XCPB	XCP	CH
-2.	-0.64	0.43	-0.18	-0.12	-0.17	-0.07	-0.09	0.02	-0.18	-0.17	0.09	0.00	-0.65	0.41	17.78	4.42	5.38	10.52	-0.19	
0.	0.00	0.41	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.41	17.78	4.42	-0.00	8.01	0.00	
2.	0.64	0.43	0.18	0.12	0.17	0.07	0.09	0.02	0.18	0.17	0.00	0.00	0.65	0.41	17.78	4.42	3.38	10.52	0.19	
4.	1.48	0.52	0.49	0.24	0.15	0.25	0.11	0.49	0.35	0.00	0.00	1.50	0.41	17.78	4.42	5.06	11.10	-0.43		
6.	2.43	0.71	0.34	0.37	0.53	0.22	0.47	0.30	0.85	0.54	0.00	0.00	2.47	0.45	17.78	4.42	6.08	11.33	-1.25	
8.	3.48	1.03	1.22	0.48	0.72	0.29	0.76	0.62	1.23	0.72	0.00	0.00	3.53	0.54	17.78	4.42	6.77	11.43	-2.17	
10.	4.62	1.52	1.62	0.60	0.92	0.36	1.12	1.11	1.62	0.92	0.00	0.00	4.68	0.69	17.78	4.42	7.27	11.49	-3.13	
12.	5.83	2.20	2.02	0.71	1.11	0.43	1.55	1.79	2.03	1.11	0.00	0.00	5.91	0.94	17.78	4.42	7.66	11.51	-4.10	
14.	7.12	3.10	2.44	0.83	1.31	0.49	2.05	2.69	2.45	1.30	0.00	0.00	7.22	1.28	17.78	4.42	7.96	11.52	-5.06	
16.	8.48	4.26	2.86	0.93	1.50	0.56	2.63	3.85	2.87	1.50	0.00	0.00	8.61	1.75	17.78	4.42	8.22	11.52	-5.99	

CDINL=0.0

CDRAFT=0.0

CDPROT=0.0

CDNIR=0.3036 CDOW=0.0736 CNOT=0.0321 CDMSC=0.0292 CDNWT=0.4092

VW¹= 1.50 DELTA= 10.00

	AL	CLTOT	CNTOT	CLNP	CLNW	CLTP	CLR	CLRT	CLR	CNI	CNIP	CNTP	CNTD	CNTN	CN	CA	XCPW	XCP	XCP	XCP	CM
-2.	0.19	0.43	-0.12	-0.12	0.28	0.24	-0.09	0.00	-0.12	0.28	0.51	0.02	0.18	0.44	17.78	4.42	3.38	****	4.29		
0..	0.92	0.46	0.00	0.00	0.61	0.31	0.00	0.04	0.00	0.61	0.51	0.02	0.92	0.46	17.78	4.42	-0.00	4.42	5.86		
2.	1.52	0.53	0.12	0.12	0.81	0.38	0.09	0.10	0.12	0.81	0.51	0.02	1.53	0.48	17.78	4.42	3.38	6.46	6.69		
4.	2.60	0.69	0.44	0.24	1.02	0.45	0.25	0.27	0.44	1.02	0.50	0.02	2.43	0.52	17.78	4.42	5.06	8.28	6.17		
6.	3.32	0.97	0.80	0.37	1.23	0.52	0.47	0.54	0.81	1.24	0.50	0.02	3.43	0.61	17.78	4.42	6.08	9.24	5.44		
8.	4.47	1.39	1.19	0.48	1.44	0.59	0.76	0.97	1.20	1.45	0.49	0.02	4.53	0.76	17.78	4.42	6.77	9.80	4.61		
10.	5.63	1.99	1.59	0.60	1.66	0.66	1.12	1.12	1.56	1.60	1.67	0.49	0.02	5.70	0.98	17.78	4.42	7.27	10.16	3.74	
12.	6.87	2.79	2.00	0.71	1.87	0.72	1.55	2.36	2.01	1.89	0.48	0.01	6.95	1.30	17.78	4.42	7.66	10.41	2.84		
14.	8.17	3.81	2.42	0.83	2.08	0.78	2.05	3.39	2.43	2.10	0.47	0.01	8.28	1.72	17.78	4.42	7.96	10.59	1.93		
16.	9.54	5.10	2.85	0.93	2.29	0.84	2.63	4.68	2.85	2.30	0.47	0.01	9.67	2.27	17.78	4.42	8.22	10.71	1.03		

CDINL=0.0

CDAFT=0.0

CDPROT=0.0

CDOB=0.3036

CDON=0.0736

CDOT=0.0321

CDMISC=0.0292

CDONBT=0.4092

VXH= 1.50 DELTA= 20.00

	AL	CLTOT	CNTOT	CLNP	CLRW	CLTP	CLBT	CLR	CDI	CNIP	CNTP	CLTD	CNTD	CN	CA	XCPW	XCP	XCPB	XCP	CM
-2.	0.76	0.50	-0.05	-0.12	0.51	0.52	-0.09	0.03	-0.05	0.51	0.97	0.06	0.75	0.52	17.78	4.42	3.38	1.39	7.06	
0.	1.89	0.62	0.00	0.00	1.31	0.58	0.00	0.16	0.00	1.31	0.96	0.06	1.89	0.62	17.78	4.42	-0.00	4.42	12.13	
2.	2.44	0.73	0.05	0.12	1.53	0.65	0.09	0.26	0.05	1.54	0.95	0.06	2.46	0.64	17.78	4.42	3.38	5.34	13.49	
4.	3.32	0.96	0.36	0.24	1.75	0.71	0.25	0.49	0.37	1.76	0.93	0.06	3.35	0.73	17.78	4.42	5.06	6.92	13.10	
6.	4.33	1.33	0.75	0.37	1.97	0.97	0.78	0.47	0.86	0.76	1.99	0.92	0.05	4.39	0.87	17.78	4.42	6.08	8.03	12.22
8.	5.42	1.84	1.15	0.48	2.19	0.84	0.76	1.38	1.16	2.21	0.90	0.05	5.49	1.07	17.78	4.42	6.77	8.76	11.31	
10.	6.58	2.53	1.56	0.60	2.40	0.90	1.12	2.07	1.57	2.43	0.88	0.05	6.67	1.35	17.78	4.42	7.27	9.27	10.35	
12.	7.81	3.42	1.98	0.71	2.60	0.95	1.55	2.96	1.99	2.64	0.87	0.05	7.92	1.72	17.78	4.42	7.66	9.64	9.34	
14.	9.10	4.54	2.40	0.83	2.80	1.01	2.05	4.09	2.41	2.85	0.85	0.05	9.23	2.21	17.78	4.42	7.96	9.92	8.30	
16.	10.44	5.92	2.83	0.93	2.99	1.06	2.63	5.47	2.84	3.04	0.83	0.04	10.61	2.81	17.78	4.42	8.22	10.14	7.23	

CDINL=0.0 CDAFT=0.0 CDPROT=0.0

CDOB=0.3036 CDOB=0.0736 CDT=0.0321 CDMSC=0.0292 CDOMBT=0.4092

VXH= 1.50 DELTA= 30.00

	AL	CLTOT	CDTOT	CLWV	CLWP	CLTP	CLUT	CLB	CDI	CHWP	CNTP	CLTD	CDTD	CK	CA	XCPW	XCP	XCPB	XCP	CM
-2.	0.91	0.56	-0.13	-0.12	0.53	0.73	-0.09	0.04	-0.14	0.52	1.32	0.11	0.89	0.59	17.78	4.42	3.38	0.62	9.08	
0.	2.77	0.85	0.00	0.00	1.99	0.79	0.00	0.34	0.00	1.99	1.23	0.11	2.77	0.85	17.78	4.42	-0.00	4.42	17.75	
2.	3.38	1.01	0.13	0.12	2.20	0.84	0.09	0.50	0.14	2.21	1.26	0.10	3.41	0.90	17.78	4.42	3.38	5.41	18.45	
4.	4.04	1.23	0.24	0.24	2.40	0.90	0.25	0.73	0.25	2.42	1.24	0.10	4.08	0.95	17.78	4.42	5.06	6.08	19.34	
6.	5.07	1.67	0.68	0.37	2.60	0.95	0.47	1.17	0.69	2.64	1.21	0.09	5.14	1.13	17.78	4.42	6.08	7.32	17.98	
8.	6.15	2.24	1.10	0.48	2.79	1.00	0.76	1.75	1.11	2.84	1.18	0.09	6.24	1.37	17.78	4.42	6.77	8.14	16.73	
10.	7.28	2.99	1.52	0.60	2.98	1.06	1.12	2.49	1.53	3.04	1.14	0.08	7.39	1.68	17.78	4.42	7.27	8.73	15.47	
12.	8.47	3.93	1.95	0.71	3.15	1.10	1.55	3.44	1.96	3.22	1.11	0.08	8.61	2.08	17.78	4.42	7.66	9.18	14.16	
14.	9.72	5.08	2.38	0.83	3.31	1.15	2.05	4.60	2.39	3.39	1.07	0.07	9.88	2.58	17.78	4.42	7.96	9.53	12.79	
16.	11.02	6.48	2.81	0.93	3.46	1.19	2.63	6.01	2.82	3.55	1.04	0.07	11.22	3.20	17.78	4.42	8.22	9.81	11.37	

CNIML=0.0 CNIAFT=0.0 CNPROT=0.0

CN01=0.3036 CN0W=0.0736 CNOT=0.0321 CMIS=0.0292 CNMBT=0.4092

VXR4 = 2.00 DELTA = 0.0

	AL	CLTOT	CNTOT	CLWIP	CLWIP	CLSHW	CLTP	CLWT	CLW	CDI	CNTP	CNWIP	CNTD	CNTD	CNTD	CNTD	CN	CA	XCPW	XCPB	XCP	CM	CM
-2.	-0.60	0.41	-0.18	-0.11	-0.16	-0.06	-0.09	0.02	-0.18	-0.16	0.00	0.00	-0.61	0.00	-0.61	0.00	0.39	17.78	4.42	3.38	10.50	-0.20	
0.	0.00	0.39	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.39	17.78	4.42	-0.00	8.15	0.00	
2.	0.60	0.41	0.18	0.11	0.16	0.06	0.09	0.02	0.18	0.16	0.00	0.00	0.00	0.61	0.39	17.78	4.42	3.38	10.50	0.20	0.20	0.53	
4.	1.39	0.49	0.48	0.22	0.33	0.13	0.25	0.10	0.48	0.33	0.00	0.00	1.42	0.39	17.78	4.42	5.07	11.05	-0.53	0.53	0.53	0.53	
6.	2.30	0.67	0.81	0.32	0.50	0.19	0.48	0.27	0.82	0.50	0.00	0.00	2.34	0.42	17.78	4.42	6.11	11.27	-1.04	1.04	1.04	1.04	
8.	3.41	1.42	1.18	0.43	0.53	0.68	0.25	0.78	1.19	1.19	0.00	0.00	3.36	0.50	17.78	4.42	6.81	11.37	-1.84	1.84	1.84	1.84	
10.	4.41	1.42	1.56	0.53	0.53	0.86	0.31	1.15	1.15	1.15	0.00	0.00	4.47	0.64	17.78	4.42	7.33	11.42	-2.68	2.68	2.68	2.68	
12.	5.60	2.08	1.95	0.63	1.04	0.37	1.61	1.68	1.96	1.96	0.00	0.00	5.68	0.87	17.78	4.42	7.73	11.44	-3.51	3.51	3.51	3.51	
14.	6.89	2.96	2.35	0.73	1.23	0.43	2.15	2.57	2.36	2.23	0.00	0.00	6.99	1.21	17.78	4.42	8.05	11.44	-4.32	4.32	4.32	4.32	
16.	8.27	4.13	2.76	0.82	1.42	0.48	2.79	3.74	2.76	1.41	0.00	0.00	8.40	1.69	17.78	4.42	8.32	11.42	-5.08	5.08	5.08	5.08	

CDONLT=0, CDONRT=0, CDONST=0, CDONUT=0, CDONWT=0, CDONXT=0

VXM= 2.00 DELTA= 10.00

	AL	CLTOT	CLTP	CLRP	CLST	CLR	CNI	CNIP	CNTP	CNTD	CNTD	CN	CA	XCPW	XCP	CH1.	
-2.	0.16	0.41	-0.12	-0.11	0.26	0.23	-0.09	0.00	-0.13	0.26	0.48	0.01	0.15	0.42	17.78	4.42	3.38 **** 3.90
0.	0.86	0.44	0.00	0.00	0.57	0.29	0.00	0.03	0.00	0.57	0.48	0.01	0.86	0.44	17.78	4.42	-0.00 4.50 5.43
2.	1.44	0.50	0.12	0.11	0.76	0.35	0.09	0.09	0.13	0.77	0.47	0.01	1.45	0.45	17.78	4.42	3.38 6.21 6.54
4.	2.27	0.65	0.43	0.22	0.96	0.41	0.25	0.24	0.44	0.97	0.47	0.01	2.30	0.49	17.78	4.42	5.07 8.30 5.79
6.	3.21	0.90	0.73	0.32	1.36	0.47	0.48	0.49	0.79	1.17	0.46	0.01	3.25	0.56	17.78	4.42	6.11 9.23 5.18
8.	4.25	1.29	1.15	0.43	1.37	0.53	0.78	0.88	1.16	1.37	0.46	0.01	4.30	0.69	17.78	4.42	6.81 9.78 4.49
10.	5.37	1.85	1.54	0.53	1.57	0.59	1.15	1.44	1.54	1.58	0.45	0.01	5.44	0.89	17.78	4.42	7.33 10.13 3.75
12.	6.59	2.61	1.93	0.63	1.77	0.64	1.61	2.21	1.94	1.78	0.45	0.01	6.67	1.19	17.78	4.42	7.73 10.37 3.00
14.	7.89	3.62	2.34	0.73	1.97	0.69	2.15	3.21	2.34	1.98	0.44	0.01	7.99	1.60	17.78	4.42	8.05 10.54 2.24
16.	9.28	4.91	2.75	0.82	2.17	0.75	2.79	4.51	2.75	2.18	0.43	0.01	9.42	2.17	17.78	4.42	8.32 10.66 1.51

CDIML=0.0 CDAFT=0.0

CDPROT=0.0
CDOB=0.2912 CDOW=0.0705 CDOT=0.0307 CDMS=0.0292 CDOB=0.3924

VXA= 2.00 DELTA= 20.00

	AL	CLTOT	CNTOT	CLWP	CLBU	CLTP	CLBT	CLB	CDI	CRWP	CHTP	CLTD	CNTD	CN	CA	XCPW	XCP	XCP	CH
-2.	0.68	0.47	-0.07	-0.11	0.46	0.49	-0.09	0.02	-0.07	0.45	0.90	0.05	0.66	0.49	17.78	4.42	3.38	1.28	6.33
0.	1.79	0.58	0.00	0.00	1.25	0.54	0.00	0.14	0.00	1.25	0.89	0.05	1.79	0.58	17.78	4.42	-0.00	4.50	11.31
2.	2.32	0.68	0.07	0.11	1.45	0.60	0.09	0.24	0.07	1.46	0.88	0.05	2.33	0.60	17.78	4.42	3.38	5.43	12.58
4.	3.15	0.89	0.37	0.22	1.66	0.65	0.25	0.45	0.37	1.67	0.87	0.05	3.18	0.67	17.78	4.42	5.07	6.96	12.29
6.	4.11	1.22	0.74	0.32	1.87	0.71	0.48	0.78	0.74	1.89	0.85	0.05	4.16	0.78	17.78	4.42	6.11	8.04	11.56
8.	5.16	1.70	1.12	0.43	2.08	0.76	0.78	1.26	1.12	2.10	0.84	0.05	5.22	0.96	17.78	4.42	6.81	8.75	10.80
10.	6.28	2.34	1.51	0.53	2.28	0.81	1.15	1.91	1.52	2.31	0.82	0.04	6.36	1.22	17.78	4.42	7.33	9.25	9.99
12.	7.49	3.19	1.91	0.63	2.48	0.86	1.61	2.76	1.92	2.51	0.81	0.04	7.59	1.57	17.78	4.42	7.73	9.62	9.13
14.	8.77	4.29	2.32	0.73	2.67	0.91	2.15	3.86	2.33	2.71	0.79	0.04	8.90	2.04	17.78	4.42	8.05	9.89	8.26
16.	10.14	5.67	2.73	0.82	2.85	0.95	2.79	5.24	2.74	2.89	0.77	0.04	10.31	2.66	17.78	4.42	8.32	10.11	7.37

CDI:HL=0.0 CDAFT=0.0 CDPROT=0.0

CDOB=0.2912 CDOU=0.0705 CDOT=0.0307 CDMSC=0.0292 CDDWB1=0.3924

VXH= 2.00 DELTA= 30.00

	AL	CLTUT	CDTUT	CLWTP	CLJIN	CLLTP	CLBT	CLB	CNI	CNMP	CNTP	CLTD	CNTD	CN	CA	XCPW	XCPT	XCPB	XCP	CM
-2.	0.79	0.52	-0.14	-0.11	0.45	0.68	0.09	0.03	-0.14	0.44	1.23	0.10	0.78	0.55	17.78	4.42	3.38	0.52	7.99	
0.	2.63	0.73	0.00	0.00	1.90	0.73	0.00	0.30	0.00	1.90	1.20	0.09	2.63	0.79	17.78	4.42	-0.00	4.49	16.64	
2.	3.22	0.94	0.14	0.11	2.10	0.78	0.09	0.45	0.14	2.11	1.18	0.09	3.24	0.82	17.78	4.42	3.38	5.47	17.33	
4.	3.85	1.14	0.26	0.22	2.29	0.83	0.25	0.66	0.27	2.31	1.15	0.09	3.89	0.87	17.78	4.42	5.07	6.15	18.18	
6.	4.83	1.53	0.67	0.52	2.49	0.87	0.48	1.06	0.68	2.52	1.12	0.08	4.89	1.02	17.78	4.42	6.11	7.34	17.01	
8.	5.86	2.06	1.07	0.43	2.67	0.92	0.78	1.60	1.08	2.71	1.09	0.08	5.94	1.23	17.78	4.42	6.81	8.14	15.94	
10.	6.96	2.76	1.48	0.53	2.84	0.96	1.15	2.30	1.48	2.90	1.06	0.07	7.06	1.51	17.78	4.42	7.33	8.72	14.85	
12.	8.13	3.66	1.89	0.63	3.01	1.00	1.61	3.20	1.89	3.07	1.03	0.07	8.26	1.89	17.78	4.42	7.73	9.16	13.70	
14.	9.37	4.79	2.30	0.73	3.16	1.03	2.15	4.34	2.31	3.23	1.00	0.06	9.53	2.38	17.78	4.42	8.05	9.51	12.51	
16.	10.70	6.20	2.72	0.82	3.30	1.07	2.79	5.75	2.72	3.38	0.97	0.06	10.89	3.01	17.78	4.42	8.32	9.78	11.29	

CNINL=0.0 CDIAFT=0.0 CNPROT=0.0

CNO3=0.2912 CNOV=0.0705 CNOT=0.0307 CNMISC=0.0292 CNOGBT=0.3924

VXH= 2.50 DELTA= 0.0

AL	CLTOT	CNTOT	CLMP	CLWV	CLTP	CLBT	CLB	CDI	CHMP	CHTP	CLTD	CDTD	CH	CA	XCPW	XCPT	XCPB	XCP	CM
-2.	-0.56	0.33	-0.13	-0.09	-0.15	-0.06	-0.09	0.02	-0.18	-0.15	0.00	0.00	-0.58	0.37	17.78	4.42	3.38	10.50	-0.19
0.	0.00	0.34	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.38	17.78	4.42	-0.00	8.27	0.00
2.	0.56	0.39	0.18	0.09	0.15	0.06	0.09	0.02	0.18	0.15	0.00	0.00	0.58	0.37	17.78	4.42	3.38	10.50	0.19
4.	1.31	0.47	0.47	0.18	0.31	0.11	0.25	0.09	0.47	0.31	0.00	0.00	1.34	0.37	17.78	4.42	5.09	11.03	-0.27
6.	2.17	0.63	0.79	0.27	0.47	0.17	0.48	0.25	0.80	0.47	0.00	0.00	2.21	0.40	17.78	4.42	6.14	11.23	-0.91
8.	3.14	0.91	1.14	0.35	0.64	0.22	0.79	0.53	1.15	0.64	0.00	0.00	3.19	0.46	17.78	4.42	6.86	11.32	-1.61
10.	4.21	1.34	1.51	0.44	0.81	0.28	1.18	0.96	1.51	0.81	0.00	0.00	4.27	0.59	17.78	4.42	7.40	11.37	-2.34
12.	5.39	1.93	1.89	0.52	0.98	0.33	1.68	1.60	1.89	0.98	0.00	0.00	5.47	0.81	17.78	4.42	7.82	11.38	-3.06
14.	6.70	2.87	2.27	0.60	1.16	0.38	2.29	2.49	2.28	1.15	0.00	0.00	6.80	1.16	17.78	4.42	8.16	11.37	-3.73
16.	8.14	4.09	2.67	0.68	1.34	0.43	3.03	3.71	2.67	1.33	0.00	0.00	8.28	1.69	17.78	4.42	8.44	11.34	-4.32

CNTUL=0.0 CNAFT=0.0 CNPROT=0.0

CNOR=0.2796 CNOW=0.0682 CNOT=0.0297 CNMIS=0.0296 CNOWBT=0.3776

VIM= 2.50 DFLTA= 10.00

	AL	CLTOT	CNTOT	CLWMP	CLBRI	CLTP	CLUT	CLR	CNI	CNHP	CNTD	CNTO	CLTD	CNTN	CH	CA	XCPW	XCPY	XCPZ	CM
-2.	0.14	0.39	-0.13	-0.09	0.24	0.21	-0.09	0.00	-0.13	0.24	0.45	0.01	0.13	0.40	17.78	4.42	3.38	****	3.56	
0.	0.81	0.42	0.00	0.00	0.54	0.27	0.00	0.03	0.00	0.54	0.44	0.01	0.81	0.42	17.78	4.42	-0.00	4.58	5.04	
2.	1.36	0.47	0.13	0.09	0.72	0.52	0.09	0.08	0.15	0.72	0.44	0.01	1.37	0.42	17.78	4.42	3.38	6.60	5.77	
4.	2.14	0.60	0.43	0.18	0.91	0.38	0.25	0.21	0.43	0.91	0.44	0.01	2.17	0.45	17.78	4.42	5.09	8.31	5.43	
6.	3.04	0.84	0.76	0.27	1.10	0.43	0.48	0.45	0.77	1.11	0.43	0.01	3.08	0.51	17.78	4.42	6.14	9.22	4.92	
8.	4.03	1.20	1.12	0.35	1.29	0.48	0.79	0.81	1.12	1.30	0.43	0.01	4.08	0.62	17.78	4.42	6.86	9.76	4.33	
10.	5.13	1.73	1.43	0.44	1.49	0.53	1.18	1.34	1.43	1.50	0.42	0.01	5.19	0.81	17.78	4.42	7.40	10.11	3.70	
12.	6.33	2.47	1.87	0.52	1.68	0.58	1.68	2.08	1.87	1.69	0.42	0.01	6.41	1.10	17.78	4.42	7.82	10.34	3.06	
14.	7.65	3.47	2.26	0.60	1.87	0.63	2.29	3.08	2.26	1.88	0.41	0.01	7.76	1.52	17.78	4.42	8.16	10.50	2.45	
16.	9.10	4.82	2.66	0.68	2.06	0.67	3.03	4.43	2.66	2.07	0.40	0.01	9.25	2.13	17.78	4.42	8.44	10.62	1.89	

CDINL=0.0 CDAFT=0.0 CDPROT=0.0

CDOW=0.2796 CDOW=0.0682 CNOT=0.0297 CNOT=0.0296 CDOWST=0.3776

	AL	CLTOT	CLHP	CLHW	CLTP	CLRT	CLR	CDI	CHWP	CNTP	CLTD	CTD	CN	CA	XCPW	XCP	XCPB	XCP	CH
-2.	0.61	0.44	-0.08	-0.09	0.41	0.46	-0.09	0.02	-0.08	0.41	0.84	0.05	0.59	0.46	17.78	4.42	3.38	1.18	5.70
0.	1.69	0.55	0.00	0.00	1.19	0.51	0.00	0.13	0.00	1.19	0.83	0.04	1.69	0.55	17.78	4.42	-0.00	4.57	10.59
2.	2.20	0.63	0.08	0.09	1.39	0.56	0.09	0.21	0.08	1.39	0.82	0.04	2.22	0.56	17.78	4.42	3.38	5.51	11.77
4.	2.98	0.82	0.37	0.18	1.59	0.60	0.25	0.40	0.37	1.59	0.81	0.04	3.02	0.61	17.78	4.42	5.09	6.99	11.55
6.	3.90	1.13	0.72	0.27	1.78	0.65	0.48	0.71	0.73	1.80	0.80	0.04	3.95	0.71	17.78	4.42	6.14	8.05	10.94
8.	4.91	1.57	1.08	0.35	1.98	0.70	0.79	1.15	1.09	2.00	0.78	0.04	4.97	0.87	17.78	4.42	6.86	8.75	10.29
10.	6.00	2.18	1.46	0.44	2.18	0.74	1.18	1.76	1.47	2.20	0.77	0.04	6.08	1.10	17.78	4.42	7.40	9.24	9.60
12.	7.20	3.00	1.85	0.52	2.36	0.78	1.68	2.59	1.85	2.39	0.75	0.04	7.29	1.44	17.78	4.42	7.82	9.60	8.88
14.	8.50	4.10	2.24	0.60	2.55	0.82	2.29	3.68	2.25	2.58	0.74	0.03	8.63	1.92	17.78	4.42	8.16	9.88	8.15
16.	9.93	5.53	2.64	0.68	2.72	0.86	3.03	5.12	2.64	2.75	0.72	0.03	10.10	2.58	17.78	4.42	8.44	10.08	7.45

CD1HL=0.0 CDAFT=0.0 CDPROT=0.0
 CD0B=0.2796 CDDW=0.0682 CDOT=0.0297 CDMSC=0.0296 CDMBT=0.3776

VMIN= 2.50 . DELTA= 50.00
 AL CLTOT CLTOT CLWP CLWP CLB CLB CDI CDI CHTP CHTP CTD CTD CN CA XCPN XCPN XCPB XCPB XCP XCP CMA
 -2. 0.69 0.43 -0.14 -0.09 0.38 0.64 -0.09 0.02 -0.15 0.37 1.14 0.08 0.68 0.51 17.78 4.42 3.38 0.43 7.03
 0. 2.50 0.73 0.00 0.00 1.82 0.68 0.00 0.27 0.00 1.82 1.12 0.08 2.50 0.73 17.78 4.42 -0.00 4.55 15.67
 2. 3.06 0.87 0.14 0.09 2.01 0.72 0.09 0.41 0.15 2.02 1.10 0.08 3.08 0.76 17.78 4.42 3.38 5.52 16.33
 4. 3.66 1.05 0.27 0.18 2.20 0.77 0.25 0.60 0.28 2.21 1.07 0.07 3.70 0.79 17.78 4.42 5.09 6.20 17.12
 6. 4.60 1.41 0.67 0.27 2.38 0.80 0.48 0.97 0.67 2.41 1.05 0.07 4.66 0.93 17.78 4.42 6.14 7.36 16.11
 8. 5.59 1.90 1.05 0.35 2.55 0.84 0.79 1.46 1.05 2.59 1.02 0.07 5.66 1.11 17.78 4.42 6.86 8.14 15.18
 10. 6.66 2.56 1.43 0.44 2.72 0.88 1.18 2.12 1.44 2.77 0.99 0.06 6.75 1.37 17.78 4.42 7.40 8.71 14.23
 12. 7.82 3.43 1.83 0.52 2.88 0.91 1.68 3.00 1.83 2.93 0.96 0.06 7.94 1.73 17.78 4.42 7.82 9.15 13.23
 14. 9.08 4.56 2.23 0.60 3.02 0.94 2.29 4.13 2.23 3.08 0.93 0.06 9.23 2.23 17.78 4.42 8.16 9.50 12.21
 16. 10.47 6.03 2.63 0.68 3.16 0.97 3.03 5.60 2.63 3.22 0.90 0.05 10.66 2.91 17.78 4.42 8.44 9.77 11.18

CDINL=0.0 CDRAFT=0.0 CNPROT=0.0
 CN03=0.2796 CDOW=0.0682 CDOT=0.0297 CMISCS=0.0296 CMORT=0.3776

VXH= 3.00 DELTA= 10.00

	AL	CLTOT	CNTOT	CLNP	CLRW	CLRT	CLTP	CNI	CNR	CNTP	CNTP	CNTW	CA	XCPW	XCP	CP1	
-2.	0.12	0.38	-0.13	-0.07	0.22	0.20	-0.09	0.00	-0.14	0.22	0.42	0.01	0.11	0.38	17.78	4.42	3.38 ****
0.	0.76	0.40	0.00	0.00	0.51	0.25	0.00	0.03	0.00	0.51	0.41	0.01	0.76	0.40	17.78	4.42	0.00
2.	1.28	0.45	0.13	0.07	0.68	0.30	0.09	0.07	0.14	0.68	0.41	0.01	1.29	0.40	17.78	4.42	3.38
4.	2.03	0.57	0.42	0.15	0.86	0.35	0.25	0.19	0.42	0.86	0.41	0.01	2.05	0.43	17.78	4.42	5.10
6.	2.39	0.78	0.74	0.22	1.04	0.40	0.49	0.41	0.74	1.05	0.40	0.01	2.92	0.48	17.78	4.42	8.34
8.	3.85	1.12	1.08	0.29	1.23	0.44	0.81	0.75	1.09	1.23	0.40	0.01	3.90	0.57	17.78	4.42	5.09
10.	4.93	1.93	1.44	0.57	1.41	0.49	1.23	1.26	1.44	1.42	0.39	0.01	4.93	0.75	17.78	4.42	6.66
12.	6.14	2.37	1.81	0.43	1.60	0.53	1.77	1.99	1.81	1.60	0.39	0.01	6.22	1.04	17.78	4.42	9.23
14.	7.50	3.41	2.18	0.50	1.78	0.57	2.47	3.03	2.18	1.78	0.38	0.01	7.62	1.49	17.78	4.42	4.16
16.	9.88	5.78	2.57	0.57	1.96	0.61	4.18	5.41	2.56	1.96	0.38	0.01	10.07	2.84	17.78	4.42	2.66
																3.01	

CDINL=0.0 CNAFT=0.0 CDPROT=0.0

CDDBB=0.2688 CDDBV=0.0664 CDOT=0.0289 CDMSC=0.0301 CDONBT=0.3642

VIN= 3.00 DELTA= 20.00

AI	CLTOT	CNTOT	CLUP	CLDN	CLTP	CLBT	CLB	CNI	CLNP	CNTP	CLTN	CNTN	CN	CA	XCPW	XCP	XCP	CM	
-2.	0.53	0.42	-0.09	-0.07	0.36	0.43	-0.09	0.01	-0.09	0.36	0.79	0.04	0.52	0.44	17.78	4.42	3.38	0.97	5.10
0.	1.60	0.51	0.00	0.00	1.13	0.47	0.00	0.11	0.00	1.13	0.78	0.04	1.60	0.51	17.78	4.42	-0.00	4.63	9.90
2.	2.03	0.59	0.09	0.07	1.32	0.52	0.09	0.19	0.09	1.32	0.77	0.04	2.10	0.52	17.78	4.42	3.38	5.59	11.00
4.	2.84	0.77	0.37	0.15	1.51	0.56	0.25	0.37	0.37	0.37	1.52	0.75	0.04	2.87	0.57	17.78	4.42	5.10	10.85
6.	3.71	1.05	0.70	0.22	1.70	0.60	0.49	0.65	0.71	1.71	0.74	0.04	3.76	0.65	17.78	4.42	6.17	8.07	10.34
8.	4.69	1.46	1.05	0.29	1.89	0.81	0.64	0.81	1.06	1.91	0.73	0.03	4.75	0.79	17.78	4.42	6.92	8.76	9.80
10.	5.76	2.04	1.42	0.37	2.07	0.68	1.23	1.65	1.42	2.10	0.72	0.03	5.84	1.01	17.78	4.42	7.48	9.24	9.22
12.	6.97	2.86	1.79	0.43	2.25	0.72	1.77	2.47	1.79	2.28	0.70	0.03	7.06	1.35	17.78	4.42	7.93	9.60	8.63
14.	8.32	3.99	2.17	0.50	2.43	0.76	2.47	3.60	2.17	2.45	0.68	0.03	8.45	1.86	17.78	4.42	8.29	9.87	8.07
16.	10.68	6.52	2.56	0.57	2.59	0.79	4.18	6.13	2.55	2.62	0.67	0.03	10.89	3.32	17.78	4.42	8.86	10.06	8.28

CDIML=0.0 CDRAFT=0.0 CDPROT=0.0

CDOM=0.2688 CDOM=0.0664 CDOT=0.0289 CDMISC=0.0301 CDOWN=0.3642

VXW= 3.00 DELTA= 30.00

	AL	CLTOT	CHTOT	CLWP	CLBW	CLTP	CLBT	CLB	CNI	CHWP	CNTP	CLTD	CDTD	CN	CA	XCPN	XCPPT	XCPB	XCP	CM
-2.	0.59	0.45	-0.15	-0.07	0.31	0.60	-0.09	0.02	-0.15	0.31	1.07	0.07	0.58	0.47	17.78	4.42	3.38	0.20	6.14	
0.	2.38	0.68	0.00	0.00	1.74	0.64	0.00	0.25	0.00	1.74	1.05	0.07	2.38	0.68	17.78	4.42	-0.00	4.61	14.75	
2.	2.91	0.80	0.15	0.07	1.92	0.67	0.09	0.37	0.15	1.93	1.02	0.07	2.93	0.70	17.78	4.42	3.38	5.57	15.38	
4.	3.50	0.98	0.28	0.15	2.10	0.71	0.25	0.55	0.29	2.12	1.00	0.06	3.53	0.73	17.78	4.42	5.10	6.26	16.12	
6.	4.39	1.31	0.66	0.22	2.28	0.74	0.49	0.88	0.66	2.30	0.98	0.06	4.44	0.84	17.78	4.42	6.17	7.38	15.26	
8.	5.34	1.77	1.02	0.29	2.44	0.78	0.81	1.35	1.03	2.48	0.95	0.06	5.42	1.01	17.78	4.42	6.92	8.15	14.45	
10.	6.40	2.40	1.39	0.37	2.60	0.81	1.23	1.98	1.40	2.65	0.92	0.05	6.49	1.25	17.78	4.42	7.48	8.72	13.63	
12.	7.57	3.26	1.77	0.43	2.75	0.84	1.77	2.85	1.77	2.80	0.90	0.05	7.68	1.62	17.78	4.42	7.93	9.16	12.78	
14.	8.88	4.43	2.15	0.50	2.89	0.87	2.47	4.02	2.15	2.94	0.87	0.05	9.03	2.15	17.78	4.42	8.29	9.50	11.93	
16.	11.20	7.03	2.54	0.57	3.02	0.89	4.18	6.62	2.54	3.07	0.84	0.05	11.43	3.67	17.78	4.42	8.86	9.78	11.84	

CDTTL=0.0 CDAFT=0.0 CDPROT=0.0

CD03=0.2688 CD01=0.0664 CD0T=0.0289 CNMISCC=0.0301 CD0WBT=0.3642

X. CONCLUSIONS AND RECOMMENDATIONS

This thesis has presented the methods and general procedures for the conceptual design of tactical missiles. As mentioned, this is not necessarily the best design. Now that the design procedure has been attempted once, the reason for this is readily seen. The process is one of continuous compromise. As was shown in Chapter 7, the optimum wing for lift is not the best wing for minimizing drag. We also saw in Chapter 8 that increasing chamber pressure increases thrust at the expense of increased weight. This thesis has tried to point out some of these areas of compromise and present methods to deal with them.

Areas which were not covered which need to be investigated in conceptual design are structures, radar cross-section and cost. With the increased emphasis on survivability and the decreasing budget, it becomes increasingly important to define the effects of these areas on design early in the process.

The complexity of the design process and the need to obtain timely and accurate information have made it ideally suited for the digital computer. The AEROFC program used in this thesis is part of a large scale computer program (MISSYN) which consists of modules for each section of the design analysis.

With a good understanding of the theory and methods used in missile design, the computer aided design program with graphics capability gives the designer the capacity to make intelligent

design iterations and almost instantaneously see the effect of the change on all areas of design. A limited example of this can be seen in the use of AERO1. A change in the performance requirements for maneuver capability would require a redesign of the lifting surfaces. A change in the lifting surface design would change the drag characteristics of the missile and therefore, the propulsion requirements. The AERO1 program coupled with a similar propulsion module would allow the designer to make the changes and instantly see the penalty or savings in propellant weight.

One pass at the design has been accomplished in this thesis. As was seen throughout the process, decisions in one area affect the design in others. For this reason the design process becomes an iterative one. The final design of the first iteration is the baseline missile for the second iteration and the design is started again. By making several passes through the loop, the solution converges on the final design.

```

SUBROUTINE GEJSUB
DIMENSION X(16),XOT(16),XAL(48)
COMMON REFT,SSUBS,XLOB,ZF,VXM,IZZY,WLL,NSURF
COMMON ICSC,CLAMW2,ISWP,IAFBT,IDUM1,
COMMON CLAMW,ICLAMW,ISNOSE,BWT,BW2,IAP,BW1,BW2,CROOT,CROOT2,
12ST,SW,SW2,CLAMW2,ISNOSE,TOVCT,XLAM1,XLAMM,XLAMW2,
3XMAC,XMAC2,XACT,XWING,XWING2,XTAIL,XTAIL2,XLAREA,XREF,
4ISWEP,ISWEPT,ISWEP2,NWING,NTAIL,NWING2,ENGINE,ENLET,BETA,DBASE,
5DJET,XLABODD,ART,ARWA,RW2,BAR,BCOLAM,BETAI,B1,CLALT,CLALW,CLALW2,CLAL1,
ICOLAM,ROOT,IAFB,ISWP,TOVCT,ODC,XBCRBW,XBCRWB,XKBW,XKBT,XKTB,XMAC,D1
2XKB,XKWB,XKWB1,ISWP1,RATIO,XLAM1,XMAC,D1
DRAT=XLNOSE/D
DRAA=XLABODD
D1=D

XLOBB=XL/D
IF(CLAMW=5.) 10,10,20
10
CLAMW=5
IF(CLAMW2=5.) 30,30,40
30
CLAMW2=5
IF(CLAM=5.) 40,50,60
40
CLAM=5
50
CONTINUE
60
CLAMW=CLAMW/57.*29578
CLAMW2=CLAMW2/57.*29578
IF((INDSE-1)) 1571,1572,1571
1572
SSI=(3.*14*DRAT)+0.4
GO TO 1579
1571
IF(INOS=2) 1573,1574,1573
1574
SSI=(2.*68*DRAT)+0.1
GO TO 1579
1573
IF(INDSE=3) 1579,1576,1579
1576
SSI=(2.*DRAT)+2
1579
SSI=3.*14159*0.4*(XL-XLNOSE-XLABODI)+(SSI+2.*68*DRAA*0.91)*(3.14159*0
1D/4.0)
1
      WRITE(6,*)'SSUBS'
      FORMAT('IX,ISNOSE,WETTED AREA = ',F10.4,/)
9111
      IF(DRAT-.6) 1580,1581
1580
      GO TO 1594
      ZF=1.522*EXP(-.7*DRAT)
1581
      Z=HT*0.001
1594
      IF(HT-.35332*0) 1595,1595,1596
1595
      T=519.-HT/1280
      PS=(1.-91-.01315*Z)*5.256
      GO TO 1597
1596
      T=393.

```



```

ISWP=ISWPW
XLAM=XLAMW
AR=ARW
IF((BAR>GT-.5*.0 AND .VXM=.GE.1.0) BAR=.5.0
505 IF((BAR>GT-.5*.0 AND .VXM.LT.1.0) BAR=.5.0
      5 IF((XLAM->.25) .5*.200
      10 IF(.VXM-<1.0) 20*.10*.10
      20 CLAR=-.1833*BAR+.1.6
      20 GO TO 370
      30 CLAR=(XLAM-.50)*40*.110*.50
      40 CLAR=.001032518*BAR**8+.0200677*BAR**7-.1557225*BAR**6
           1+.607131*BAR**5-.1*21537*BAR**4+.13786*BAR**3-.5718*BAR**2
           1+.458487*BAR+.1.57588
      160 TO 370
      110 CLAR=.00121006*BAR**8-.0254597*BAR**7+.22119793*BAR**6
           1-.020448*BAR**5+.27651997*BAR**4-.3.6809957*BAR**3
           1+.97903*BAR**2+.27640134*BAR+1.57129
      160 TO 370
      50 CLAR=-.00007776*BAR**8-.001814086*BAR**7+.051777483*BAR**6
           1-.43555555*BAR**5+.1.7240831*BAR**4-.3.3472653*BAR**3
           1+.2.6137236*BAR**2-.23389542*BAR+1.57998
      160 TO 370
      60 IF(.VXM-<1.70*.170*.80
      70 CLAR=-.1667*BAR+.1.575
      70 GO TO 370
      80 IF((BAR>2.190*.90*.100
      100 CLAR=.0013364*BAR**5--.0318237*BAR**4+.26049*BAR**3-.8643*BAR**2
           1+.669396*BAR+.2.2008
      160 TO 370
      90 CLAR=-.040277*BAR**4+.222522*BAR**3-.48931*BAR**2+.47287*BAR+.1.57
      200 GO TO 370
      220 IF((XLAM-<1.1) 210*.260*.260
      220 CLAR=-.2077*BAR+.1.575
      230 IF((BAR>2.25) 240*.240*.250
      240 CLAR=.2077*BAR+.1.575
      250 GO TO 370
      251 CLAR=-.1668*BAR+.1.667
      252 CLAR=1.587*(1.26)*(2.-BAR)
      260 GO TO 370
      270 IF((XLAM-<1.3) 270*.320*.320
      280 CLAR=-.2065*BAR+.1.6
      280 GO TO 370

```

AD-A095 118

NAVAL POSTGRADUATE SCHOOL MONTEREY CA
TACTICAL MISSILE CONCEPTUAL DESIGN, (U)
SEP 80 D R REDMON

F/6 16/4.1

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4 of 4
AD-A095 118

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290 CLAR=002119027*BAR**5-03203282*BAR+1.550048
1-61820362*BAR**2+.62703849*BAR+1.550048
GO TO 370 AERO1450
AERO1460
AERO1470
AERO1480
AERO1490
AERO1500
AERO1510
AERO1520
AERO1530
AERO1540
AERO1550
AERO1560
AERO1570
AERO1580
AERO1590
AERO1600
AERO1610
AERO1620
AERO1630
AERO1640
AERO1650
AERO1660
AERO1670
AERO1680
AERO1690
AERO1700
AERO1710
AERO1720
AERO1730
AERO1740
AERO1750
AERO1760
AERO1770
AERO1780
AERO1790
AERO1810
AERO1820
AERO1830
AERO1840
AERO1850
AERO1860
AERO1870
AERO1880
AERO1890
AERO1900
AERO1910
AERO1920

320 IF(VXM-1*1 330,330,340
330 CLAR=-2*5*BAR+1.675
GO TO 370
1-3:3445976*BAR**3+2.2919403*BAR**2-21045286*BAR+1.6203524
1-2.025071933*BAR**6-25587433*BAR**5+1.0834904*BAR**4
1-2.1562044*BAR**3+1.6307710*BAR**2-.084287165*BAR+1.7039
370 CLAL=CLAR#AR
IF(KFIN-2) 500,501,502
500 CLALW=CLAL
GO TO 503
501 CLALW2=CLAL
GO TO 503
502 CLALT=CLAL
503 IF(KFIN-2) 504,507,900
504 KFIN=KFIN+1
IF(SW2-1) 507,507,506
506 AR=(BW2-D)*2/SW2
BAR=BETAI*AR
ISWP=ISWPW2
ARW2=AR
XLAH=XLAH2
GO TO 505
507 KFIN=KFIN+1
IF(ST) 509,509,508
508 AR=(BT-D)**2/ST
BAR=BETAI*AR
ART=AR
ISWP=ISWP1
NSURFENTAIL
XLAH=XLAH1
GO TO 505
509 IF(SW1) 510,05100,900
5100 IF(SW2) 511,05110,900
5110 IF(ST) 980,980,900
980 LLLL=2
900 LLKK=0
902 LIZZ=IZZY+1
IF(SW1) 510,901
IF(SW2) COJ(CLAW)/{SIN(CLAMW+.11)
901 COJ(CLAW=(BW-D)**2/SW
BCOLAM=BETAI*COLAM

```

```

CROOT=CROOTW
B1=BW
IAFB=IAFBW
XMAC=XMACW
TOVC=TOVCW
CLAL1=CLALW
XCLAM1=XCLAMW
ISWP1=ISWPW
NSURF=NWING
BAR=BETAI*ARW
RATIO=CROOT/(BETAI*D)
LLLL=0
RETURN
510 IZZY = IZZY + 1
942 IF (SW2) 942,942,511
      LLLL=1
      RETURN
511 COLAM=COS(CLAMW2)/(SIN(CLAMW2+.1))
      BCOLAM=BETAI*COLAM
      CROOT=CROOTW2
      B1=BW2
      IAFB=IAFBW2
      CLAL1=CLALW2
      XCLAM1=XCLAMW2
      NSURF=NWING2
      XMAC=XMACW2
      TOVC=TOVCW2
      ISWP1=ISWPW2
      ARW2=(BW2-D)**2/SW2
      BAR=BETAI*ARW2
      RATIO=CROOT/(BETAI*D)
      LLLL=0
      RETURN
END

```

```

      SUBROUTINE CATSUB(XMM(16),XDT(16),XAL(48))
      DIMENSION XMM(16),XDT(16),XAL(48)
      COMMON REFT,SSUBS,XLOB,ZF,YXM,
      COMMON ICSC,INODE,W2,IABW,IACB,IACW,IACR,IACM,
      COMMON CLAMN,CLANW,CLAN2,DCLN,NOSE,BT,BW,CROOT,
      COMMON SW,SW2,T0VCH,T0VCW,T0VCT,XLAMN,XLAMW,XLAMW2,
      COMMON 2ST,3XHACH2,XHACH2,XACT,XWING,XWING2,XTAIL,
      COMMON 4ISWEPT,ISWEP2,NWING,NTAIL,ENGINE,ENLET,BETA,DBASE,
      COMMON ARW2,BAR,BCOLAM,BETAL,BLACLA,T,CLALW2,CLAL1,
      COMMON ART,ARW2,ARW2,ARW2,ARW2,ARW2,ARW2,ARW2,ARW2,
      COMMON 1COLAM,CROOT,IAFBT,IADUM1,IADUM2,IADUM2,
      COMMON AERO2300,AERO2310,AERO2320,AERO2330,AERO2340,AERO2350,AERO2360,AERO2370,AERO2380,AERO2390,AERO2400.

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XKB2=BCOLAM**2*RATIO**2*ARCOS(1./BCOLAM)-BCOLAM*RATIO**2*SQRT(BCOLA) AERO02890
1AH**2-1.*ARSIN(1./RATIO) AERO02900
XKB3=SQRT(BCOLAM**2-1.*ALOG(RATIO+SQRT(RATIO**2-1.)) AERO02910
XKBW=((8./(3.14159*SQRT(BCOLAM**2-1.))*BETAI*CLAL1)*(XLAM1+1.)*(BL/DAER02920
1-1.*1.*((1./RATIO)*(XKB1-XKB2-XKB3)) AERO02930
1 GO TO 1000 AERO02940
9295 IF(RATIO-1.) 9296,9312,9312 AERO02950
9296 RATIO=1. D=CROOT/BETAI AERO02960
12 XKBI=(1.+COLAM*BETAI)*SQRT((RATIO-1.)*(COLAM*BETAI*RATIO+1.)) AERO02970
9312 D=AERO02980
1 XKB2=RATIO**2*(BCOLAM*RATIO**2*(BCOLAM+1.)*(ATAN(SQRT(BCOLAM+1.)/(COLAM*BETAI*RATIO+1.))) AERO02990
1 (1./BCOLAM)-ATAN(SQRT((RATIO-1.)/(COLAM*BETAI*RATIO+1.))) AERO03000
1 XKB3=((BCOLAM+1.)/SQRT(BCOLAM))*0.5*(ALOG(1.+SQRT((BCOLAM*(RATIO-1.)*(COLAM*(RATIO-1.)/(COLAM* AERO03010
1.)/(COLAM*BETAI*RATIO+1.))-ALOG(1.-SQRT((BCOLAM*(RATIO-1.)/(COLAM* AERO03020
2*BETAI*RATIO+1.))) AERO03030
1 XKBW=((16.*SQRT(BCOLAM)*(1./RATIO)/(BETAI*CLAL1)*(XLAM1+1.)*(BL/DI AERO03040
1-1.*3.*14159*(BCOLAM+1.)))*(XKB1-XKB2-XKB3)) AERO03050
1000 XKBT=XKBW-XKTB AERO03060
D=D AERO03070
1 IF(XKB-1.0) 2000,2010 AERO03080
2000 IF(1SWP1-1.) 2020,2020,2030 AERO03090
2020 IF(BAR-2.0) 2040,2050,2050 AERO03100
2040 XBCRWB=0.35-XLAM1*(0.35+17.885-SQRT(328.8782-(BAR-3.)*2)) AERO03110
2050 GO TO 2080 AERO03120
2050 XBCRWB=0.35-0.1*XLAM1 AERO03130
2050 GO TO 2080 AERO03140
2070 XBCRWB=0.55-0.3*XLAM1 AERO03150
2070 GO TO 2110 AERO03160
2080 IF(BAR-2.0) 2060,2070,2070 AERO03170
2080 XCR1=SQRT(2029.5+(BAR-3.)*2) AERO03180
2080 XBCRWB=(-44.5*XCR1)-XLAM1*(-44.5*XCR1)+17.885-SQRT(328.8782-(BAR AERO03190
1-3.)*2)) AERO03200
1 GO TO 2110 AERO03210
2090 XBCRBW=0.25-XLAM1*(32.125-SQRT((1032.02-(BAR-4.)*2))+((1.-XLAM1)* AERO03220
1 ALOG(1.-0.1*0./B1)*(72.25-(BAR-4.0)*2)) AERO03230
2100 XBCRBW=0.25+((1.-XLAM1)*ALOG(1.04+0.1*D/B1)) AERO03240
2100 GO TO 2400 AERO03250
2110 IF(BAR-4.0) 2120,2130,2130 AERO03260
2120 XBCRBW=0.25*((1.-2.*XLAM1)*(32.125-SQRT((1032.02-(BAR-4.)*2)))+ AERO03270
1 ((1.-XLAM1)*ALOG(1.12+0.3*D/B1)*(-7.5+SQRT(72.25-(BAR-4.)*2))) AERO03280
2130 XBCRBW=0.25+((1.-XLAM1)*ALOG(1.12+0.3*D/B1)) AERO03290
2130 GO TO 2400 AERO03300
2140 IF(1SWP1-1. 2160,2170,2170 AERO03310
2140 IF(BAR-3.0) 2164,2174,2174 AERO03320
2140 IF(1SWP1-1. 2160,2170,2170 AERO03330
2140 IF(BAR-3.0) 2164,2174,2174 AERO03340
2140 IF(1SWP1-1. 2160,2170,2170 AERO03350
2140 IF(BAR-3.0) 2164,2174,2174 AERO03360

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```

2160 XBCRWB=-{(9.235+25.*(1.-XLAM1))+SQRT((9.71+25.*(1.-XLAM1))**2-
1 BAR-3.1**2}
1 GO TO 200
2170 XBCRWB=0.005*BAR+0.46
2180 XBCRWB=0.675-XLAM1*(0.675+9.235-SQRT(94.1-(BAR-3.1)**2))
2190 XBCRWB=0.005*BAR+0.46+0.2*(1.-XLAM1)
2200 GO TO 2300
2200 BARLAM=BAR*((1.-XLAM1)*(1.+t(1./BCOLAM)))
2200 IF(BARLAM-4.-0)2210,2220
2210 IF(BAR-2.0)2230,2240
2230 XBMID1=ALOG((1.32-0.32*XLAM1)
2235 XBMID2=4.+(XBMID1**2)-(0.5+0.5139*(D/B1)*(1.17+XLAM1)*(1./(0.331
1*(D/B1)))**2
1*(D/B1))1
1 XBMID=XBMID2/XBMID3
1 XBMID=4.+((XBMID1-XBMID)**2
1 XBCRBW=SQRT(XCMBID-((BAR-2.1)**2))+XB MID
2240 XBCRBW=0.5+0.25695*(D/B1)*(1./(0.331**((D/B1)))*BAR*(1.17+XLAM1)
2220 GO TO 2400
2220 IF((AFB)2250,2250,2280,2280,2270
2250 IF(RATIO-1.0)2280,0.67
2280 XBCRBW=0.67
2270 XBCRBW=-2.32+SQRT(8.9401-((1./RATION)-1.)***2)
2260 GO TO 2400
2260 XBCRBW=(0.429/RAT10)+0.5
2300 GO TO 2400
2300 BARLAM=BAR*((1.-XLAM1)*(1.+((1./BCOLAM)))
2310 IF(BARLAM-4.-0)2310,2220
2310 IF(BAR-2.0)2320,2240
2320 XBMID1=ALOG((1.65-0.654*XLAM1)
2320 GO TO 2235
2400 D=D1
2400 ARAT=BAR/BETAI
2400 IF(ARAT-4.0)20,20,21
21 ARAT=4.0
20 ODC=2.*((1.+XLAM1)**1.6*EXP(-.4*ARAT))
21 IF(ARAT-1.0)22,23,23
23 IF(XLAM1.LE.0.25)GO TO 27
23 IF(XLAM1.LE.0.75)GO TO 28
23 ODC=ODC-.25*(ARAT-1.)
23 IF(ARAT.LT.2.0)GO TO 22
23 ODC=(2.*((1.+XLAM1)**1.6)*.45-(.85*(ARAT-2.))

```

```

27 GO TO 22
28 GO TO 22
29 GO TO 22
30 CONTINUE
31 RETURN
32 END

```

```

3248 FORMAT( 6X, SHISWPT, 5X, SHIAFBT, 10X, SHXLANT, 10X, SHXTAIL, / )
3241 READ 110X5H BT, 9X6HCROOT, 10X5H ST, 10X5H XMACI, 10X5H XMACII, 10X5H XMACIII, 5X, 5HNDLT, 5X, 5HNMACH, 5X, 5HNALPH,
3241 FORMAT( //, 6X, 5HICSC, 5X, 5HINSE, 5X, 5HDELT, 5X, 5HNMACI, 5X, 5HNALPH, AERO4340
3241 FORMAT( //, 5HNBODY ), AERO4350
3241 FORMAT( //, 3F10.3, 10 3F6.1, 2 ), AERO4360
3241 FORMAT( //, 3F10.3, 10 3F6.1, 2 ), AERO4370
3334 FORMAT( //, 5X, 0ENGINE, 5X, ENLET, 5X, BETAI, 5X, DBASE, 5X, DJET, AERO4380
3335 FORMAT( //, 5X, 0CDPROF ), AERO4390
3336 WRITE( 6, 300, 10 ), AERO4400
3333 WRITE( 6, 300, 10 ), AERO4410
30010 FORMAT( //, 1X, INPUT TITLE ), AERO4420
1 READ( 110, TITL0, TITL01, TITL02, TITL03, TITL04, TITL05, TITL6, TITL7, TITL8, TITL9, AERO4430
1112 FORMAT( //, 12 ), AERO4440
1115 FORMAT( //, 10, 3 ), AERO4450
60001 WRITE( 6, 100, 10 ), AERO4460
10010 FORMAT( //, 1X, INPUT CONTROL CONSTANT, 1=TAIL CONTROL, 2=WING CONTROL ), AERO4470
1 READ( 1112, 13=CANARD, TITL0, TITL1, TITL2, TITL3, TITL4, TITL5, TITL6, TITL7, TITL8, TITL9, AERO4480
1 IF( IABC.EQ.1 ) READ( 1112, ICSC ), AERO4490
1 IF( IABC.EQ.1 ) READ( 1112, IDT ), AERO4500
1 IF( IABC.EQ.1 ) READ( 1112, 100, 20 ), AERO4510
10002 WRITE( 6, 100, 20 ), AERO4520
10020 FORMAT( //, 1X, INPUT NOSE SHAPE, 1=ELLIPSOID, 2=OGIVE, AERO4530
1 READ( 1112, 13=NONE ), AERO4540
1 IF( IABC.EQ.1 ) READ( 1112, INOSE ), AERO4550
1 IF( IABC.EQ.1 ) READ( 1112, 100, 30 ), AERO4560
10003 FORMAT( //, 1X, INPUT NUMBER OF CONTROL DEFLECTIONS ), AERO4570
1 READ( 1112, IDT ), AERO4580
1 IF( IABC.EQ.1 ) READ( 1112, 100, 40 ), AERO4590
1 IF( IABC.EQ.1 ) READ( 1112, 100, 40 ), AERO4600
10004 FORMAT( //, 1X, INPUT NUMBER OF MACH NUMBERS ), AERO4610
1 READ( 1112, IM ), AERO4620
1 IF( IABC.EQ.1 ) READ( 1112, 100, 50 ), AERO4630
1 IF( IABC.EQ.1 ) READ( 1112, 100, 50 ), AERO4640
10005 FORMAT( //, 1X, INPUT NUMBER OF ANGLES OF ATTACK ), AERO4650
1 READ( 1112, IAT ), AERO4660
1 IF( IABC.EQ.1 ) READ( 1112, 100, 60 ), AERO4670
1 IF( IABC.EQ.1 ) READ( 1112, 100, 60 ), AERO4680
10006 WRITE( 6, 100, 60 ), AERO4690
10060 READ( 1112, NBODY ), AERO4700
1 IF( IABC.EQ.1 ) READ( 1112, 100, 70 ), AERO4710
10007 WRITE( 6, 100, 70 ), AERO4720
10070 READ( 1112, ISWPW ), AERO4730
1 IF( IABC.EQ.1 ) READ( 1112, 100, 80 ), AERO4740
10008 WRITE( 6, 100, 80 ), AERO4750
10080 FORMAT( //, 1X, 0=NO BODY AFTER WING, 1=BODY AFTER WING ), AERO4760
1 READ( 1112, IAFBW ), AERO4770
1 IF( IABC.EQ.1 ) READ( 1112, 100, 90 ), AERO4780
10090 IF( IABC.EQ.1 ) READ( 1112, 100, 90 ), AERO4790
10080 IF( IABC.EQ.1 ) READ( 1112, 100, 90 ), AERO4800

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60009 WRITE(6,10090) INPUT WING SWEET CONSTANT*)
10090 FORMAT(6,15001)
      WRITE(6,1X,IF DELTA PLANFORM ENTER 0, OTHERWISE*)
55001 FORMAT(6,1X,13)
      WRITE(6,1X,1=UNSWEPT LEADING EDGE*)
1113 FORMAT(6,1X,1=SWEPT LEADING EDGE*)
1114 READ 1112 ISWEPW TO 55000
      IF(1ABC-EQ-1)GO TO 55000
60010 WRITE(6,10100)
      READ 1112 NWING TO 55000
      IF(1ABC-EQ-1)INPUT NUMBER OF WINGS*
10100 FORMAT(6,1X,1SWPT
      READ 1112 NWING
      IF(1ABC-EQ-1)GO TO 55000
60011 WRITE(6,10110)
      READ 1112 ISWPT
      IF(1ABC-EQ-1)GO TO 55000
10110 FORMAT(6,1X,1NO BODY AFTER TAIL, 1=NON-DELTA TAIL, 2=DELTA TAIL*)
      WRITE(6,10120)
      READ 1112 IAFBT
      IF(1ABC-EQ-1)GO TO 55000
10120 FORMAT(6,1X,1NO BODY AFTER TAIL, 1=BODY AFTER TAIL*)
      READ 1112 IAFBT
      IF(1ABC-EQ-1)GO TO 55000
60012 WRITE(6,10130)
      READ 1112 ICLAMW
      IF(1ABC-EQ-1)GO TO 55000
10130 FORMAT(6,1X,1INPUT TAIL SWEET CONSTANT*)
      WRITE(6,10130)
      WRITE(6,1113)
      WRITE(6,1114)
      READ 1112 ISWEPT
      IF(1ABC-EQ-1)GO TO 55000
60014 WRITE(6,10140)
      READ 1112 NTAIL
      IF(1ABC-EQ-1)GO TO 55000
      INPUT NUMBER OF TAILS*)
10140 FORMAT(6,1X,1INPUT NUMBER OF TAILS*)
      WRITE(6,10150)
      READ 1115 XLAMW
      IF(1ABC-EQ-1)GO TO 55000
      INPUT TIP-TO-CHORD RATIO OF WING*)
10150 FORMAT(6,1X,1INPUT TIP-TO-CHORD RATIO OF WING*)
      READ 1115 XLAMW
      IF(1ABC-EQ-1)GO TO 55000
      INPUT LEADING EDGE SWEET OF WING,(DEGS)*
10160 FORMAT(6,1X,1CLAMW
      READ 1115 CLAMW
      IF(1ABC-EQ-1)GO TO 55000
60015 WRITE(6,10170)
      READ 1115 BW
      IF(1ABC-EQ-1)GO TO 55000
      INPUT WING SPAN, INCLUDE BODY*)
10170 FORMAT(6,1X,1BW
      READ 1115 BW
      IF(1ABC-EQ-1)GO TO 55000
60016 WRITE(6,10180)
      READ 1115 CROOTW
      IF(1ABC-EQ-1)GO TO 55000
      INPUT WING ROOT CHORD(AT BODY JUNCTION*)
10180 FORMAT(6,1X,1CROOTW
      READ 1115 CROOTW
      AERO4810
      AERO4820
      AERO4830
      AERO4840
      AERO4850
      AERO4860
      AERO4870
      AERO4880
      AERO4890
      AERO4900
      AERO4910
      AERO4920
      AERO4930
      AERO4940
      AERO4950
      AERO4960
      AERO4970
      AERO4980
      AERO4990
      AERO5000
      AERO5010
      AERO5020
      AERO5030
      AERO5040
      AERO5050
      AERO5060
      AERO5070
      AERO5080
      AERO5090
      AERO5100
      AERO5110
      AERO5120
      AERO5130
      AERO5140
      AERO5150
      AERO5160
      AERO5170
      AERO5180
      AERO5190
      AERO5200
      AERO5210
      AERO5220
      AERO5230
      AERO5240
      AERO5250
      AERO5260
      AERO5270
      AERO5280

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60019 IF(IABC.EQ.1)GO TO 55000
10190 WRITE(6,10190) INPUT EXPOSED WING AREA (2 PANELS)
READ 1115'S'W
1F(IABC.EQ.1)GO TO 55000
10200 WRITE(6,10200) INPUT WING MEAN GEOMETRIC CHORD
READ 1115'XMACW
1F(IABC.EQ.1)GO TO 55000
60021 WRITE(6,10210) INPUT DISTANCE FROM NOSE TO WING LE
READ 1115'XWING
1F(IABC.EQ.1)GO TO 55000
60022 WRITE(6,10220) INPUT WING T/C
10220 FORMAT(1$1X,1INPUT WING T/C)
1F(IABC.EQ.1)GO TO 55000
60023 WRITE(6,10230) INPUT TIP-TO-ROOT CHORD RATIO OF TAIL
READ 1115'XLANT
1F(IABC.EQ.1)GO TO 55000
60024 WRITE(6,10240) INPUT TAIL LEADING EDGE SWEET (DEGS)
10240 FORMAT(1$1X,1INPUT TAIL LEADING EDGE SWEET (DEGS))
1F(IABC.EQ.1)GO TO 55000
60025 WRITE(6,10250) INPUT TAIL SPAN. INCLUDING BODY
READ 1115'B
1F(IABC.EQ.1)GO TO 55000
60026 WRITE(6,10260) INPUT TAIL ROOT CHORD
10260 FORMAT(1$1X,1INPUT TAIL ROOT CHORD)
1F(IABC.EQ.1)GO TO 55000
60027 WRITE(6,10270) INPUT EXPOSED TAIL AREA(2 PANELS)
READ 1115'S'T
1F(IABC.EQ.1)GO TO 55000
60028 WRITE(6,10280) INPUT TAIL MEAN GEOMETRIC CHORD
READ 1115'XMACT
1F(IABC.EQ.1)GO TO 55000
60029 WRITE(6,10290) INPUT DISTANCE FROM NOSE TO TAIL LE
10290 FORMAT(1$1X,1TAIL
1F(IABC.EQ.1)GO TO 55000
60030 WRITE(6,10300) INPUT TAIL T/C
10300 FORMAT(1$1X,1INPUT TAIL T/C)
READ 1115'TOYCT

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AERO5770
AERO5780
AERO5790
AERO5800
AERO5810
AERO5820
AERO5830
AERO5840
AERO5850
AERO5860
AERO5870
AERO5880
AERO5890
AERO5900
AERO5910
AERO5920
AERO5930
AERO5940
AERO5950
AERO5960
AERO5970
AERO5980
AERO5990
AERO6000
AERO6010
AERO6020
AERO6030
AERO6040
AERO6050
AERO6060
AERO6070
AERO6080
AERO6090
AERO6100
AERO6120
AERO6130
AERO6140
AERO6150
AERO6160
AERO6170
AERO6180
AERO6190
AERO6200
AERO6210
AERO6220
AERO6230
AERO6240

60031      IF((IABC.EQ.1)GO TO 55000
10310      WRITE(6,11X,'INPUT ALTITUDE')
FORMAT(115,HT)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10320      FORMAT(6,103,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10330      WRITE(6,103,30)
FORMAT(115,11X,'INPUT MISSILE LENGTH')
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10340      FORMAT(6,103,40)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10350      WRITE(6,103,50)
FORMAT(115,11X,'INPUT CG LOCATION (FROM NOSE)')
READ(115,110,XCG
IF((IABC.EQ.1)GO TO 55000
10360      FORMAT(6,103,60)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10370      FORMAT(6,103,70)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10380      FORMAT(6,103,80)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10390      FORMAT(6,103,90)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10400      FORMAT(6,104,00)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10410      FORMAT(6,104,10)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10420      FORMAT(6,104,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10430      INPUT NOZZLE EXIT DIAMETER)
READ(115,110)

IF((IABC.EQ.1)GO TO 55000
10310      FORMAT(115,HT)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10320      FORMAT(6,103,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10330      WRITE(6,103,30)
FORMAT(115,11X,'INPUT MISSILE LENGTH')
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10340      FORMAT(6,103,40)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10350      WRITE(6,103,50)
FORMAT(115,11X,'INPUT CG LOCATION (FROM NOSE)')
READ(115,110,XCG
IF((IABC.EQ.1)GO TO 55000
10360      FORMAT(6,103,60)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10370      FORMAT(6,103,70)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10380      FORMAT(6,103,80)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10390      FORMAT(6,103,90)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10400      FORMAT(6,104,00)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10410      FORMAT(6,104,10)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10420      FORMAT(6,104,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10430      INPUT NOZZLE EXIT DIAMETER)
READ(115,110)

IF((IABC.EQ.1)GO TO 55000
10310      FORMAT(115,HT)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10320      FORMAT(6,103,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10330      WRITE(6,103,30)
FORMAT(115,11X,'INPUT MISSILE LENGTH')
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10340      FORMAT(6,103,40)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10350      WRITE(6,103,50)
FORMAT(115,11X,'INPUT CG LOCATION (FROM NOSE)')
READ(115,110,XCG
IF((IABC.EQ.1)GO TO 55000
10360      FORMAT(6,103,60)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10370      FORMAT(6,103,70)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10380      FORMAT(6,103,80)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10390      FORMAT(6,103,90)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10400      FORMAT(6,104,00)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10410      FORMAT(6,104,10)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10420      FORMAT(6,104,20)
READ(115,110)
IF((IABC.EQ.1)GO TO 55000
10430      INPUT NOZZLE EXIT DIAMETER)
READ(115,110)

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60043 IF(IABC-EQ-1)GO TO 55000
      WRITE(6,10430)FORMAT('1X',INPUT BOAT TAIL LENGTH')
10430 READ(1115,IABODD)
      IF(IABC-EQ-1)GO TO 55000
      WRITE(6,10440)FORMAT('1X',INPUT PROTUBERANCE DRAG)
10440 READ(1115,CPROT)
      IF(IABC-EQ-1)GO TO 55000
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,TITL6,TITL7,TITL8,TITL9,AERO6250
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6260
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6270
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6280
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6290
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6300
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6310
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6320
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6330
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6340
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6350
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6360
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6370
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6380
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6390
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6400
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6410
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6420
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6430
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6440
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6450
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6460
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6470
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6480
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6490
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6500
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6510
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6520
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6530
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6540
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6550
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6560
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6570
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6580
      PRINT 1110,TITL01,TITL02,TITL03,TITL04,TITL05,AERO6590
      READ 11112,IANS
      IF(IANS-EQ-0)GO TO 70000
      55000 WRITE(6,55990)FORMAT('1X',IS INPUT DATA CORRECT?, 0=YES, 1=NO)
      59990 FORMAT('1X',IANS)
      READ 11112,IANS
      IF(IANS-EQ-0)GO TO 70000
      60000 FORMAT('1X',ENTER NUMBER OF VARIABLE TO BE CHANGED)
      READ 11112,IVAR
      GO TO 160010,60011,60012,60013,60014,60015,60016,60017,60018,60019,
      160010,60011,60012,60013,60014,60015,60016,60017,60018,60019,
      1260020,60021,60022,60023,60024,60025,60026,60027,60028,60029,
      360030,60031,60032,60033,60034,60035,60036,60037,60038,60039,
      460040,60041,60042,60043,60044),IVAR
      70000 CONTINUE
      PIE=3.14159
      IL=1+IL
      LLKK=0
      LLLL=0
      XCG2=XCG
      IZZY=0
      IF(INOSE.EQ.3160 TO 9961

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DELT A=DELT A1/57.29578+.000000001
DO 6000 J=1,AL
AL=ALPHA/57.29578+.00000001
SINAAL=SIN(AL)
COSAAL=COS(AL)
VXMR1=VXN
IZZY=IZZY+1
IF (IZZY - 4) 6666,6666,1111
6666 VXM=.6
1111 CALL CLASUB
IF(LLL-1) 900,942,980
900 IF(IZZY-4) 6009,6009,925
925 CALL CATSUB((.5*(B1-D1)*1./COLAM+.25*XROOT-.25*CROOT)/
6009 XLAN14=ATAN((.5*(B1-D1))
RE=REFT*VXM*XMAC
IF (RE-1.E06) 6010,6020,6020
6010 AA=.0835
XNN=-.211
GO TO 6070
6020 IF (RE-1.E07) 6030,6040,6040
6030 AA=.052
XNN=-.177
GO TO 6070
6040 IF (RE-1.E08) 6050,6060,6060
6050 AA=.0333
XNN=-.1488
GO TO 6070
6060 AA=.0221
XNN=-.127
6070 CF=AA*RE*XNN
SURF=FLDAT(NSURF)
CDO=SURF#CF*(1+2.*TOVC+100.*TOVC**4.)
IF (IZZY-4) 6091,6091,6092
IF (IZZY-3) 6093,6094,6095
6091 IF (AL) 2401,2402,2402
6092 ODC=-ODC
2401 LLKK=LLKK+1
2402 LLKK=LLKK+1
IF (LLKK-2) 2403,2404,2420
2403 IF (SW) 2410,2410,2420
2410 LLKK=LLKK+1
IF (SW2) 2411,2411,2420
2411 LLKK=LLKK+1
2420 IF (LLKK-2) 930,943,950
930 XKWBW=XKWBW
XKBWW=XKBWW
IF (ISWEREQ.0) GO TO 963
SHIFT=TAN(CLAW)*(BW-D1)/4.0

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GO TO 965
SHIFT=0.0
XCPWB=XWINING+XBCRWB*CROOT+SHIFT
XCPBW=XWINING+XBCRBW*CROOT
ODCW=ODC(AL)*(X<WBH*XKBWW)*CLALW*SW*COS(AL)/AREA
CLWB=SIN(AL)*XKBW*CLALW*SW*COS(AL)/AREA
CLBW=CLW-CLWB
CLVISW=(SIN(AL)*SIN(AL)*SW*COS(AL)/AREA)*ODCW
CLW=CLW+CLVISH
CLWP=CLWB+CLVISH
CDOW=CDO*(ISW)/AREA
XLAMW4=XLAM14
TOVCW=TOVC
SWTOT=SW
IZZY=IZZY+1
IF(SW2)942,942,51
511 CDOLAM=COS(CLAMW2)/SIN(CLAMW2)
CDOW2=0
BCOLAM=BETA1*COLAM
CROOT=CR00W2
B1=BW2
IAFB=IAFBW2
CLAL1=CLALW2
XLAM1=XLAMW2
TOVC=TOVCW2
XMAC=XMACW2
ISWP1=ISWPW2
BAR=BETA1*ARW2
RATIO=CROOT/(BETA1*D1
IF(IZZY-4)6009,6009,925
XKBWW2=XKBW
XKBWW2=XKBW
ISWEP2:EQ.0 GO TO 948
SHIFT=TAN(CLAMW2)*(BW2-D)/4.0
GO TO 949
SHIFT=0
XCPWB2=XWINING2+XBCRWB*CROOT+SHIFT
XCPBW2=XWINING2+XBCRBW*CROOT
ODCW2=ODC(AL)*(XKBWW2+XKBWW2)*CLALW2*SW2*COS(AL)/AREA
CLWB2=SIN(AL)*XKBW2*CLALW2*SW2*COS(AL)/AREA
CLBW2=CLW2-CLWB2
CLV1W2=(SIN(AL)*SIN(AL)*SW2*COS(AL)/AREA)*ODCW2
CLW2=CLW2+(CLV1W2
CDOW2=CDO*(ISW2)/AREA
XLAM24=XLAM14
SWTOT=SW2

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942 IZZY = IZZY +1
      LLKK=LLK+2
      LF((ST) 980, 980, 940
      CLAM=COS(CLAM)*SIN(CLAM)
      ART=(BT-D)*2/ST
      BCOLAM=BETA1*COLAM
      CROOT=CROOT
      BL=BT
      BAR=BETA1*ART
      CLALI=CLALT
      IAFB=IAFBT
      XMAC=XMACT
      TOVC=TOVCT
      ISWP1=ISWPT
      XLAM=XLAM
      RATIO=CROOT/(BETA1*D)
      IF((IZZY-4) 6009,6009,925
      XKWBT=XXKBW
      XKWBT=XXTAIL+XBCRBW+CROOT
      ODC=ODC
      CLT=((XKWBT+XKBWT)*SIN(AL))*CLALI*ST*COS(AL)/AREA
      CLTB=SIN(AL)*XKWBT*CLALI*ST*COS(AL)/AREA
      CLBT=CLT-CLTB
      CLTDB=(XKTB+XKB)*SIN(DELTA)*ST*COS(AL+DELTA)/AREA
      CLBDT=CLTDB-CLTD
      CLBT=CLBT+CLBDT
      CLV1ST=((SIN(AL+DELTA)*SIN(AL+DELTA))*ST*COS(AL+DELTA)/AREA)*ODCT
      CLTP=CLTB+CLV1ST+CLTD
      CDOT=CDOT*(ST)/AREA
      STOT=ST
      XLAMT4=XLAM14
      IF((IZZY-4) 1610, 1610, 6098
      ISWEPT=EQ.0) GO TO 6097
      SHIFT=TAN(CLAM)*(BT-D)/4.0
      GOTO 6099
      SHIFT=0.0
      6099 XCPTB=XTAIL+((XKWBT*SIN(AL)*XKTB*SIN(DELTA))*XBCRW*B*SHIFT)
      1(XKWBT*SIN(AL)+XKTB*SIN(DELTA))*CROOT+SHIFT
      6098 IF((IZZY - 4) 1610, 1610, 6098
      1710 XLOB=XL/D
      ZXM=VXM*ABS(SIN(AL))
      IF((ZX4-8) 1310, 1350, 1350
      1310 CDC=2.4*SQRT((1.5129*ZXM*ZXM)
      GO TO 1391

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1CDPROT
1IF(ENG1NE.EQ.1.0) GO TO 617
CDINL=.038*CDDB
CDINT=.620
CDINL=.05*CDDB
619 CDINT=.620
617 CDINL=.0
620 DELCPD=0.
      DELCPD=(DBASE**2-DJET**2)/(D**2)
      DRAT01=(DJET**2)/(DBASE*D)
      DBDM=DBASE/D
      DELBAS=-1532+(*0247*BETAI)-(002632*(BETAI**2))
      CPBS01=-11905+(*00017*DBDM)-(0283*(DBDM**2))
      CPBS02=-0273+(*00425*DBDM)-(1143*(DBDM**2))
      CPBS03=-0612+(*3485*DBDM)-(4254*(DBDM**2))
      CPBS04=-0789+(*5252*DBDM)-(7434*(DBDM**2))
      CPBS05=-085+(*324*DBDM)-(1.407*(DBDM**2))
      IF(BETA-.3.0)770770772
      CPBS0=(CPBS01+.15)*BETA/3.0)-.15
770 GO TO 800
772 IF(BETA-.5.6)773773*(CPBS02)*(BETA-.3.01/2.6)+CPBS01
773 CPBS0=(CPBS02-CPBS02)*(BETA-3.01/2.6)+CPBS01
775 IF(BETA-.8.0)776776778
776 CPBS0=(CPBS03-CPBS02)*(BETA-5.6)/2.4)+CPBS02
778 IF(BETA-.1.6.0)779779780
779 CPBS0=(CPBS04-CPBS03)*(BETA-.8.01/8.0)+CPBS03
780 CPBS0=(CPBS05-CPBS04)*(BETA-1.6.01/4.0)+CPBS04
800 CDBASE=(-DRAT01)*(CPBS0+DELCP0+(DELCP0*DRAT01))
      IF(ENG1NE.EQ.1.0)GO TO 618
      IF(DBDM-.761801,801,802
801 CBOAT5=.1013*DBDM
802 IF(DBDM.LT.0.82) GO TO 803
      CBOAT5=-.4*(DBDM-.82)+.077
      GO TO 810
803 CBOAT5=.077
810 IF(DBDM-.245)811,811,812
811 CBOAT4=.0653*DBDM
812 GO TO 820
      IF(DBDM.GT.0.76) GO TO 813
      CBOAT4=.1048*(DBDM-.245)+.016
      GO TO 820

```

813 IF(DBDM GT; 0.82) GO TO 814
 CBOAT4=.07
 GO TO 820
 CBOAT4=-.369*(DBDM-.82)+.07
 814 CBOAT3=.0292*DBDM
 820 CBOAT3=.0292*DBDM
 821 GO TO 830
 1F(DBDM GT; 0.76) GO TO 823
 822 CDOAT3=.10125*(DBDM-.36)+.0105
 GO TO 830
 1F(DBDM GT; 0.81) GO TO 824
 823 CBOAT3=.057
 GO TO 830
 CBOAT3=-.2579*(DBDM-.81)+.051
 824 CBOAT2=.0125*DBDM
 830 CBOAT2=.0125*DBDM
 831 GO TO 840
 1F(DBDM GT; 0.5) GO TO 833
 832 CBOAT2=.0333*(DBDM-.32)+.004
 GO TO 840
 1F(DBDM GT; 0.72) GO TO 834
 833 CBOAT2=.1068*(DBDM-.5)+.01
 GO TO 840
 1F(DBDM GT; 0.761 GO TO 835
 834 CBOAT2=.0335
 GO TO 840
 CBOAT2=-.1354*(DBDM-.76)+.0335
 835 CBOAT2=-.1354*(DBDM-.76)+.0335
 840 1F(DBDM -.61841.841.842
 CBOAT1=.0055-(.03375*DBDM)+(.06875*(DBDM**2))
 841 GO TO 850
 1F(DBDM GT; 0.75) GO TO 843
 CBOAT1=.1*(DBDM-.6)+.01
 GO TO 850
 1F(DBDM GT; 0.78) GO TO 844
 CBOAT1=.025
 GO TO 850
 CBOAT1=-.091*(DBDM-.78)+.025
 844 1F(BETA-.3*.851.851.852
 CBOAT1=(CBOAT1*BETA)/3.0
 GO TO 875
 1F(BETA-.5*.853.853.854
 CBOAT1=((CBOAT2-CBOAT1)*(BETA-3.0)/2.6)+CBOAT1
 GO TO 875
 1F(BETA-.8*.855.855.856
 CBOAT1=((CBOAT3-CBOAT2)*(BETA-5.6)/2.4)+CBOAT2
 GO TO 875
 1F(BETA-.16.0)857.857.858
 CBOAT1=((CBOAT4-CBOAT3)*(BETA-8.0)/8.0)+CBOAT3

```

858 GO TO 875
859 IF(BETA-16.0)>860*(BETA-16.0)/8.0)+CBOAT4
860 PRINT 875
861 PRINT 866
862 FORMAT(1X,'BOATTAIL ANGLE GREATER THAN 24 DEGREES')
863 CBOAT1=CBOAT1+DELBAS*(DELCP0)*(DRATO1)
864 CDAFT=CBOAT1+CDBASE
865 CDBQAT=CBOAT1+DELBAS*(DELCP0)*(DRATO1)
866 CDAFT=CBOAT+CDBASE
867 CDBXLNOSE=0
868 CDBXLNOSE/0
869 CDWN1=0.000407*(FR**8.0)-0.0102*(FR**7.0)+.108*(FR**6.0)-.616*(FR**5.)+2.074*(FR**4.0)-4.183*(FR**3.0)+4.891*(FR**2.0)-3.017*
870 2FR+.7795
871 IF((FR-GT.1.0)) CDWN1=.7432093E-04*FR**5.0-.15368398E-02*FR**4.0
872 1+.01741209*FR**3.0-.039555503*FR**2.0+.050635882*FR+.005431397
873 CDPTR=(CDWN1/.8)*VXM
874 CDQB1=CDOB
875 CDQB2=CDOB+CDINL+CDAFT+CDPTR
876 CDQWB1=CDDW+CDDWBT+CDOB
877 CDQWBT=CDDWBT*.1
878 CDQWBT=CDDWBT+CDMISC
879 IF((IZY-4)) 1618.1619
880 1618 VXM=1.999999
881 IF((SW)) 1620.1621
882 1620 CDOU=0.
883 1621 IF((SW)) 1622.1623
884 1622 CDOU2=0.
885 1623 IF((ST)) 1624.1624
886 1624 CDOT=0.
887 1625 HONST=CDDWBT
888 GO TO 1639
889 1639 IF((SW)) 1700.1700.1701
890 DCOOSW=0.0
891 SWTOT=0.0
892 CDOU=0.0
893 GO TO 1702
894 1701 XMH=VXM*SQRT(COS(XLAMM4))
895 SQMITC=SQRT(LABS((XXM*XXM)-1.0))/ (TOVCW**0.33333)
896 ATC=ARW#(TOVCW##.333333)
897 IZT=1
898 1640 IF((ATC-1.5)) 1641.1642.1642
899 1641 IF((VXM-1.0)) 1643.1643.1644
900 1642 FUNCT=3.3081-1.8779*SQMITC+11.0916*SQMITC*SQMITC**4
901 1643 SQMITC#*3+7.4633*SQMITC**4
902 FUNCT=FUNC#(ATC-2.0)*0.1
903 GO TO 1650
904 FUNCT=3.0+(ATC-1.5)*0.3

```

```

1641 GO TO 1650
1642 IF (ATC-.1) 1645,1645,1646
1643 IF (VXM-.1) 1647,1647,1648
1644 IF (FUNCT=2,4) 1798,SQMITC-.3,24405*SQMITC*SQMITC
1645 IF (SQMITC=0,4) 12647,12648,2648
2647 FUNCT=FUNC(T+(1.0*(ATC-.1)))
2648 FUNCT=FUNC(T+(0.72*(ATC-.1.0)))/SQMITC
GO TO 1650
1649 FUNCT=(0.25*SQMITC)+(ATC*2.3)
GO TO 1650
1650 IF (VXM-.1.) 1649,1649,1651
1651 FUNCT=1.3333*SQMITC+ATC*.18
1652 IF (IZT=2) 1652,1653,1654
1653 DCDOSSW=FUNCT*(TOVCW*#1.66667)*((COS(XLAMW4))**2.5)
1654 IF (SQMITC.GT.1.3) DCDOSSW=0.0
DCDOSSW=DCDOSSW-LT.0.
DCDOSSW=DCDOSSW=0.0
1702 IF (SW2) 1703,1703,1704
1703 DCDOSS2 = 0.0
DCDOSS2 = 0.0
SW2TOT = 0.0
DCDOSS2 = 0.0
GO TO 1705
1704 IZT = 2
VXM=SQRT(COS(XLAM24))
SQMITC=SQRT(ABS((XXM*XXM)-1.))/TOVCW2**0.333331
ATC=ARW2*(TOVCW2**.333331)
1655 DCDOSS2=FUNCT*(TOVCW*#1.66667)*((COS(XLAM24))**2.5)
1656 IF (SQMITC.GT.1.3) DCDOSS2=0.0
IF (DCDOSS2.LT.0.0) DCDOSS2=0.0
DCDOSS2=DCDOSS2
1705 DCDOST = 0.0
DCDOST = 0.0
STTOT=0.0
CDOT = 0.0
GO TO 1708
1707 IZT = 3
VXM=SQRT(COS(XLAMT4))
SQMITC=SQRT(ABS((XXM*XXM)-1.0))/TOVCWT**0.333331
ATC=ART*(TOVCWT*.333331)
1657 DCDOST=FUNCT*(TOVCWT*#1.66667)*((COS(XLAMT4))**2.5)
1658 IF (SQMITC.GT.1.3) DCDOST=0.0
IF (DCDOST.LT.0.0) DCDOST=0.0
DCDOST=DCDOST
1708 IF (VXM-.1.0) 1707,1707,1706

```

```

C 7706 IF(VXM-2.0) 7705 7708, 7708
C 7707 COVC=1.0-(0.08*VXM)
    GO TO 7709
    7705 COVC=0.92-(0.12*(VXM-1.0))
    7708 COVC = 1.31213-0 .36633*VXM+.06038*VXM**2-.00601*VXM**3+.000275*
    1 VXM**4
    7709 CDFPTR=1.02*CFBOD*COVC*SSUBS/AREA
        CDPTR=CDDBI-1.0*CFBOD*SSUBS/AREA
        IF(VXM-1.0) 1709 * 1709 1711
        1711 CDPTR=(CDP PTR/0 *
        1709 IF(FR*6T=0.00172*(FR**8)-0.00453*(FR**7)+0.050*(FR**6)-0.304*(FR**5)
            CDWN2=0.00172*(FR**8)+3*160*(FR**2)-2*391*(FR)+1*00
            1+1*096*(FR**4)-2*406*(FR**3)+3*160*(FR**2)-2*391*(FR)+1*00
            CDWN3=0.000125*(FR**8)-0.00370*(FR**7)+0.0447*(FR**6)-2*88*(FR**5)
            1+1*076*(FR**4)-2*385*(FR**3)+3.141*(FR**2)-2.529*FR+1.300
            GO TO 760
    750 CDWN2=-.33793095E-03*FR**5 .0+76402056E-02*FR**4 .0-.67397615E-01*
        1FR**3 .0+.29429971*FR**2 .0-.65782772*FR+.65397474
        CDWN3=-.36714382E-03*FR**5 .0+.88164977E-02*FR**4 .0-.83661489E-01*
        1FR**3 .0+.3974805*FR**2 .0-.9686696*FR+1.0327385
    760 IF(VXM-1.0) 1658, 1661, 1662
    1661 CDPTR=CDWN1
    1662 IF(VXM-1.0) 1655, 1665, 1666
    1665 CDPTR=CDWN2
    1655 CDPTR=(CDWN2-CDWN1)/0.2)*(VXM-0.8)+CDWN1
    1666 IF(VXM-1.0) 1667, 1668, 1664
    1668 CDPTR=CDWN3
    1667 CDPTR=(CDWN3-CDWN2)/0.2)*(VXM-1.0)+CDWN2
    1664 CDPTR=0.0
    1658 CDDB=CDPTR+CDPTR+CDINL+CDAFT+CDPTR
        CDAL2=CDDB
        CDOT=1.0*(CDOST*(STTOT/AREA)+CDOT)
        CDOW=1.0*(CDOSW*(SWTOT/AREA)+CDOW)
        CDOW2=1.0*(CDOW2+CDOS2*(SWTOT/AREA))
        CDOWBT=CDOB+CDOT+CDOW+CDOW2+CDMISC
        IF((IZZY-4) 1659, 1659, 1663
    1659 HONST=HONST+ZF*(TONST-HONST)
        VXM=VXMR1
        IZZY=IZZY + 1
    GO TO 1

```

```

1632 CDOWBT=((HONST-TONST)/{SQR((3.0)-SQR((1.2)))*SQR((VXM))+TONST
1+((HONST-TONST)/(1.-SQR((3.0))/SQR((1.2)))*
CDOT=1.*1*(DCDT*(STTOT/AREA)+CDOT)
CDOW=1.*1*(DCDW*(SWTOT/AREA)+CDOW)
CDOW2=1.*1*(DCDW2*(SWTOT/AREA))
CDOB=CDOWBT-CDOT
CDALZ=CDDB*XK2K1*SIN((2.*AL)*SIN(AL/2.)*3.14159*D*D/(4.*AREA)+ETA*
CDB=XX2K1*((SIN(AL))**3)/AREA
1CDC*AP*((SIN(AL))**3)/AREA
1CLB=XK2K1*((SIN(2.*AL))*COS(AL/2.)*COS(AL/2.)*COS(AL/2.)*COS(AL/2.)*
1A*CDC*AP*((SIN(AL))**2)*COS(AL/2.)*COS(AL/2.)*COS(AL/2.)*COS(AL/2.)*
2L)
CA5=CDOWBT
1F(AL) 1603,1604,1603
XCPB=0.0
GO TO 1605
1603 XCPB=((XC6/XREF)-(CMB/CNB))*XREF
1605 1F(SH) 1607,1607,1606
1606 1F(ST) 2973,2973,1608
1607 1F(ST) 1990,1990,2974
1608 R=D/2
1F(SC -1) 1970,1970,1971
TT=D/BT
1971 XB1=8W/2.
XB2=BT/2.
1970
HW1=70.5*CROOTT*ABS(SIN(DELTA))+(XCPWB-XTAIL-CROOTT)*ABS(SIN(AL))
XLAM1=XLAMW
GO TO 1969
XB1=BT/2.
XB2=8W/2.
TT=D/BW
HW1=-0.5*CROOTT*ABS(SIN(DELTA))+(XCPWB-XTAIL-CROOTT)*ABS(SIN(AL))
XLAM1=XLAMW
1969 FTRT=((XB2-R)/(2.*((1.-TT)**2)/(2.*((1.-TT)**2)))*ARSIN((1.-TT**2)/(1.+TT**2)))
FW=FTRT+R
F11=(FW*R**2)/((FW**2)+(HW1**2))
H1=(HW1*R**2)/((FW**2)+(HW1**2))
ZC=FW
ZD=HW1
ZLT=0.0
DO 1800 I=1,4
ZL1=((XB1-(XLAM1*R))-(ZC*(1.-XLAM1)))/(ZC-XB1)**2)+((ZD**2)*(ZC-R)**2))
ZL2=ALOG(((ZD**2)+((ZC-XB1)**2))/(ZD*(XB1-R)))
ZL3=((1.-XLAM1)/(XB1-R))/((ZC-XB1)/ZD)-ATAN((ZC-XB1)/ZD)
1R)/ZD
ZL=((ZL1*ZL2)-ZL3)
IF((I-2),1810,1820,1820

```

```

1810 ZC=-ZC
      GO TO 1850
1820 IF(I-3) 1830,1840,1850
1830 ZL=-ZL
      ZC=FI
      GO TO 1850
1840 ZL=-ZL
      ZC=-FI
      1850 CONTINUE
      IF(ICS-1) 2970,2971
2970 ART=(BT-D)*#2/S
      XBT=BT/2
      CLI=(CLALW*CLALT*XKWB1*SIN(AL)*SW*2.*ZLT*(XBT-R))/(2.*#3.14159*ART-
      CLT*AREA*(1.+XLAMT))
      CLI=CL1*COS(AL)
      CLTP=CLTP+CLI
      CLT=CLT+CLI
      CLIT=CL I
      CLIW=0
      XCPY=XCPYB
      GO TO 2972
      ARW=(BW-D)*#2/SW
      XBW=BW/2
      CLI=(CLALW*CLALT*(XKWB1*SIN(AL)+XKTBI*SIN(DELTA))*ST*2.*ZLT*(XBW-
      1RI)/(2.*#3.14159*ARW*FRT*AREA*(1.+XLAMW))
      CLI=CL1+COS(AL)
      CLNW=CLN+CLI
      CLWP=CLWP+CLI
      CLIT=0
      XCPY=XCPWB
      GO TO 2972
      CLALT=0.
      CLID=0.
      CLTD=0.
      CLBT=0.
      CLBDT=0.
      CLTB=0.
      CLTDB=0.
      CLVIST=0.
      CLIT=0.
      CLIW=0.
      CLTP=0.
      GO TO 2972

```

2974 CLW=0.
CLBW=0.
CLWB=0.
CLYT=0.
CLW2=0.
CLBW2=0.

CLVYISW=0.
CLIT=0.
CLALW=0.
CLBW=0.

2972 IF(SW2) 700,700,701
2700

CLW2=0.
CLBW2=0.
CLWB2=0.

CLALW=0.
CLBW=0.

CLYT=0.
CLW2=0.
CLBW2=0.

CLWB2=0.

CLDODRAG

C INDUCE

701 ALPH=AL

CLTOT=CLWP+CLTP+CLB+CLBW+CLBT

CLF(LARW+GE*3.01)+GO TO 9975

CLF(VXM*GE*.85)+GO TO 9975

CDI=ABS(CL*OT*TAN(ALPHA))

GOTO 9977

9975 CDI=(CLTOT)**2/(PIE*BW**2*.7/AREA)

9977 CLF(BW978) 9979, 9978

9978 AR1=BT*2/AREA

CDTD=CLTD**2/(PIE*AR1*.7)

CDT=CDOT+(CLTP**2/(PIE*AR1*.7))

9979 CLF(BW2) 9980, 9981, 9980

9980 CDW2=CDW2**2/AREA

CDW2=CDW2**2/(PIE*AR2*.7))

9981 CLF(BW1) 9982, 9983, 9982

9982 AR3=BW*#2/AREA

CDW=CDW+(CLWP**2/(PIE*AR3*.7))

C AXIAL AND NORMAL COMPONENTS

9983 CNTD=CDTD*COSAAL-CLTD*SINAAL

CNTD=CLTD*COSAAL+CDTD*SINAAL

CNTJ=CLT*COSAAL+CDT*SINAAL

CNTP=(CLTP+CLTD)*COSAAL+(CDT+CDTD)*(CDT+CDTD)*(SINAAL)

CNW=COSAAL+CDW*SINAAL

CNW=CLW*COSAAL+CDW*SINAAL

CNW=CLW2*COSAAL+CDW2*SINAAL

CNW2=CLW2*(CDOB+CDB)*COSAAL+CDW2*SINAAL

CAB=(CDB+CDOB)*COSAAL-CLW*SINAAL

CAB2=CDW2*COSAAL-CLW2*SINAAL

CAT=CDT*COSAAL-CLT*SINAAL

CA5=CAT+CAW+CAW2+CATD+CAB

GOTO 1991

1990 CN=CNB

XCP2=XCPB

IF(XAL(J).GT.90.) GO TO 9993

AER12490
AER12500
AER12520
AER12530
AER12540
AER12550
AER12560
AER12570
AER12580
AER12590
AER12600
AER12610
AER12620
AER12630
AER12640
AER12650
AER12660
AER12670
AER12680
AER12690
AER12700
AER12710
AER12720
AER12730
AER12740
AER12750
AER12760
AER12770
AER12780
AER12790
AER12800
AER12810
AER12820
AER12830
AER12840
AER12850
AER12860
AER12870
AER12880
AER12890
AER12900
AER12910
AER12920
AER12930
AER12940
AER12950
AER12960

CA=CAS*COS(AL)*COS(ALL)
 GO TO 994
 CAS=-(.5667+.893*VXM-.1727*VXM*VXM)
 C4=CAS1*COS(AL)*COS(ALL)
 CNTINUE
 CDTOT=CDB
 CNTTOT=CLB
 ALW=0.
 ALT=0.
 CDD=0.
 CDP=0.
 CDT=0.
 CNTD=0.
 CLALW=0.
 CLWT=0.
 CLWP=0.
 CLAWT=0.
 CLAW2=0.
 CLWT2=0.
 CLWP2=0.
 CNWT=0.
 CNWT2=0.
 CNW2=0.
 CNWP=0.
 CNWP2=0.
 CNWT=0.
 CNWT2=0.
 CNW2=0.
 CNWP2=0.
 XCPH=0.
 XCPT=0.
 GO TO 1992
 NBC=(CLBW+CLBT+CLBW2)*COSAL
 CNTP=CLTP*COSAAL+CDT*SINAAL
 CNWP=CLWP*COSAAL+CDW*SINAAL
 CNW2=CLW2*COSAAL+CDW2*SINAAL
 CN=CNWP+CNTP+CNB+CNW2+CNBC
 CDI=CDI+CDB
 CDTOT=CDOWBT+CDI+CDTD
 CA=CDOT*COSAAL-CLTOT*SINAAL

C TOTAL NORMAL, AXIAL, AND DRAG FORCE COEFFICIENTS
 1991

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